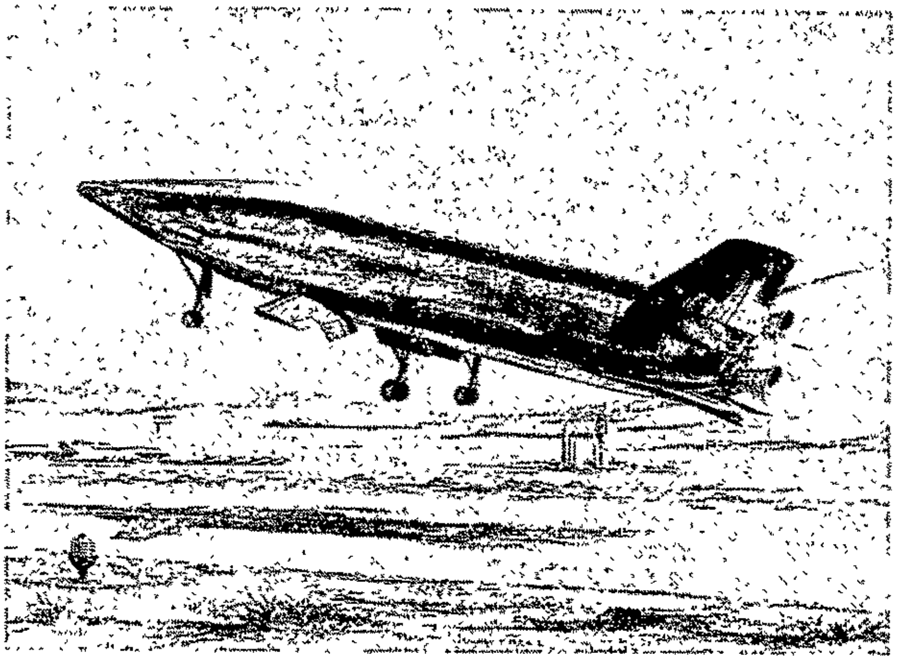
9-7
mix

CR-102537

REPORT NO. GDC-DCB69-046
CONTRACT NAS 9-9207



SPACE SHUTTLE
FINAL TECHNICAL REPORT

VOLUME V ✦ SUBSYSTEMS AND WEIGHT ANALYSIS

FACILITY FORM 602

N70-31540 (ACCESSION NUMBER)	(THRU)
446 (PAGES)	1 (CODE)
CR-102537 (NASA CR OR TMX OR AD NUMBER)	31 (CATEGORY)

GENERAL DYNAMICS
Convair Division

Reproduced by
NATIONAL TECHNICAL
INFORMATION SERVICE
US Department of Commerce
Springfield, VA. 22151

054

490

REPORT NO. GDC-DCB69-046

SPACE SHUTTLE
FINAL TECHNICAL REPORT

VOLUME V ✦ SUBSYSTEMS AND WEIGHT ANALYSIS

31 October 1969

Prepared by
CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

FOREWORD

This volume of Convair Report No. GDC-DCB 69-046 constitutes a portion of the final report for the "Study of Integral Launch and Reentry Vehicles." The study was conducted by Convair, a division of General Dynamics Corporation, for National Aeronautics and Space Administration George C. Marshall Space Flight Center under Contract NAS 9-9207 Modification 2.

The final report is published in ten volumes:

Volume I	Condensed Summary
Volume II	Final Vehicle Configurations
Volume III	Initial Vehicle Spectrum and Parametric Excursions
Volume IV	Technical Analysis and Performance
Volume V	Subsystems and Weight Analysis
Volume VI	Propulsion Analysis and Tradeoffs
Volume VII	Integrated Electronics
Volume VIII	Mission/Payload and Safety/Abort Analyses
Volume IX	Ground Turnaround Operations and Facility Requirements
Volume X	Program Development, Cost Analysis, and Technology Requirements

Convair gratefully acknowledges the cooperation of the many agencies and companies that provided technical assistance during this study:

NASA-MSFC
NASA-MSC
NASA-ERC
NASA-LaRC

Aerojet-General Corporation
Rocketdyne
Pratt and Whitney
Pan American World Airways

The study was managed and supervised by Glenn Karel, Study Manager, C. P. Plummer, Principal Configuration Designer, and Carl E. Crone, Principal Program Analyst (all of Convair) under the direction of Charles M. Akridge and Alfred J. Finzel, NASA study co-managers.

ABSTRACT

A study was made to obtain a conceptual definition of reusable space shuttle systems having multimission capability. The systems as defined can deliver 50,000-pound payloads having a diameter of 15 feet and a length of 60 feet to a 55-degree inclined orbit at an altitude of 270 n.mi. The following types of missions can be accommodated by the space shuttle system: logistics; propellant delivery; propulsive stage delivery; satellite delivery, retrieval, and maintenance; short-duration missions, and rescue missions.

Two types of reusable space shuttle systems were defined: a two-element system consisting of a boost and an orbital element and a three-element system consisting of two boost elements and an orbital element. The vehicles lift off vertically using high pressure oxygen/hydrogen rocket engines, land horizontally on conventional runways, and are fully reusable. The boost elements, after staging, perform an aerodynamic entry and fly back to the launch site using conventional airbreathing engines. Radiative thermal protection systems were defined to provide for reusability. Development programs, technology programs, schedules, and costs have been defined for planning purposes.

During the study, special emphasis was given to the following areas: System Development Approaches, Ground Turnaround Operations, Mission Interfaces and Cargo Accommodations/Handling, Propulsion System Parameters; and Integrated Electronics Systems.

TABLE OF CONTENTS

Section Page

1 INTRODUCTION. 1-1

2 GENERAL STUDY CRITERIA 2-1

3 ELECTRICAL POWER GENERATION AND DISTRIBUTION 3-1

3.1 GENERAL VEHICLE REQUIREMENTS . . 3-1
3.1.1 Design Requirements 3-1
3.1.2 Design Objectives 3-1
3.2 LOAD ANALYSIS 3-1
3.3 TRADE STUDIES 3-3
3.3.1 Batteries 3-3
3.3.2 Solar Cells 3-5
3.3.3 Fuel Cells. 3-7
3.3.4 System Selection 3-10
3.4 POWER SYSTEM DESCRIPTION 3-11
3.4.1 Orbiter. 3-11
3.4.2 Booster 3-13
3.5 FUEL CELL THERMODYNAMIC CHARACTERISTICS 3-13
3.5.1 Reactant Supply 3-15
3.5.2 Product Water Removal 3-15
3.5.3 Thermal Control 3-15
3.6 FUEL CELL ELECTRICAL CHARACTERISTICS 3-18
3.6.1 Steady State Performance 3-18
3.6.2 Short Circuit Capability 3-21
3.6.3 Transient Response 3-21
3.6.4 Type of Power 3-24
3.6.5 Buses and Feeders 3-25
3.6.6 Distribution 3-27
3.6.7 Circuit Breakers 3-27
3.7 ORBITER ELECTRICAL POWER SYSTEM WEIGHT AND VOLUME ESTIMATES . . . 3-28
3.7.1 Orbiter. 3-28
3.7.2 Booster Power System Weights. 3-30
3.8 FAILURE MODES, DETECTION AND ACTION 3-31
3.8.1 Electrical System 3-31
3.8.2 Reactant Supply 3-33

TABLE OF CONTENTS, Contd

Section Page

3.8.3 Product Water Removal 3-34
3.8.4 Thermal Control 3-34

4 AERODYNAMIC CONTROL SYSTEM 4-1

5 ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM. 5-1

5.1 GROSS SYSTEM REQUIREMENTS 5-1
5.1.1 Crew Compartment, Orbiter 5-1
5.1.2 Crew Compartment, Booster 5-1
5.1.3 General Requirements for the EC/LSS . . 5-1
5.2 TRADE STUDIES 5-3
5.2.1 Crew Compartment, Orbiter 5-3
5.2.2 Crew Compartment, Booster 5-13
5.3 DETAIL SYSTEM REQUIREMENTS . . . 5-16
5.3.1 Crew Compartment, Orbiter 5-16
5.3.2 Crew Compartment, Booster 5-20
5.4 DETAIL DESIGN DATA, ORBITER EC/LSS 5-20
5.4.1 Atmosphere Supply and Pressurization Control. 5-20
5.4.2 Atmosphere Purification 5-24
5.4.3 Water Management 5-26
5.4.4 Waste Management 5-29
5.4.5 Food Management 5-31
5.4.6 Personal Hygiene 5-31
5.4.7 Thermal Control 5-32
5.4.8 Orbiter EC/LSS Weight and Volume Summary 5-36
5.5 DETAIL DESIGN DATA, BOOSTER . . . 5-36
5.6 EXTENDED DURATION 5-36
5.6.1 Oxygen Storage for Extended Duration . . 5-38
5.6.2 CO_2 Absorber Storage for Extended Duration 5-38
5.6.3 Food for Extended Duration 5-38
5.6.4 Waste Management for Extended Duration 5-38
5.6.5 Parametric Data for Extended Duration . . 5-39

TABLE OF CONTENTS, Contd

Section Page

5.7 FAILURE MODES, DETECTION AND ACTION 5-40
5.7.1 Atmosphere Supply and Pressurization Control. 5-41
5.7.2 Atmosphere Purification 5-42
5.7.3 Water Management. 5-43
5.7.4 Waste Management. 5-44
5.7.5 Food Management 5-45
5.7.6 Personel Hygiene 5-45
5.7.7 Thermal Control 5-45
5.8 REFERENCES 5-46

6 HYDRAULIC POWER GENERATION AND DISTRIBUTION 6-1

6.1 REQUIREMENTS 6-1
6.1.1 Functions 6-1
6.1.2 Power Requirements 6-1
6.2 SYSTEM DESCRIPTION 6-1
6.3 THERMAL CONSIDERATIONS 6-8
6.4 ACTUATION SYSTEM FAILURE MODES ANALYSIS. 6-9
6.5 SYSTEM WEIGHT 6-13

7 AUXILIARY POWER UNIT 7-1

7.1 GENERAL REQUIREMENTS. 7-1
7.2 POWER AND ENERGY REQUIREMENTS . 7-1
7.3 TRADE STUDIES 7-1
7.3.1 Hydrazine Blend Fueled APU 7-2
7.3.2 H_2-O_2 FUELED APU 7-7
7.4 SYSTEM SELECTION 7-12
7.5 TOTAL SYSTEM WEIGHT ESTIMATES . . 7-12

8 THERMOSTRUCTURAL DESIGN 8-1

8.1 INTRODUCTION. 8-1
8.2 DESIGN REQUIREMENTS AND CRITERIA (T-18 CONFIGURATION) . . . 8-4
8.2.1 Design Requirements 8-4
8.2.2 Structural Design Criteria 8-9
8.2.3 Load Analysis 8-11

TABLE OF CONTENTS, Contd

Section Page

8.3 MATERIALS 8-15
8.4 SELECTED THERMOSTRUCTURAL CONCEPTS 8-20
8.4.1 General Description 8-20
8.4.2 Crew Compartment. 8-23
8.4.3 Turbofan Engine Compartment 8-23
8.4.4 Transition to LO_2 Tank 8-26
8.4.5 Liquid Oxygen Tank 8-27
8.4.6 Auxiliary Propellant Tanks 8-28
8.4.7 Liquid Hydrogen Tank. 8-32
8.4.8 Centerbody 8-33
8.4.9 Aft Body Structure 8-43
8.4.10 TPS Support Structure (FR-3 and FR-4 Orbiters and Boosters) 8-54
8.4.11 Wing Structure 8-57
8.4.12 Stabilizer Structure 8-64
8.4.13 Interstage Connections 8-67
8.4.14 Thermal Protection System 8-70
8.5 FR-3 BOOSTER THERMOSTRUCTURAL CONCEPT. 8-80
8.6 COMPOSITE MATERIALS APPLICATION 8-86
8.7 REFERENCES 8-90

9 LANDING GEAR. 9-1

9.1 REQUIREMENTS 9-1
9.1.1 General 9-1
9.1.2 Booster 9-1
9.1.3 Orbiter. 9-1
9.1.4 Common Hardware 9-2
9.2 BASELINE CONFIGURATION 9-2
9.2.1 General Description 9-2
9.2.2 Shock Absorbers 9-2
9.2.3 Wheels and Tires 9-2
9.2.4 Wheel Brakes 9-3
9.2.5 Nose Wheel Steering 9-3
9.2.6 Turnover Angle 9-4
9.2.7 Landing Gear Arrangement 9-4
9.3 SYSTEM WEIGHT 9-4

TABLE OF CONTENTS, Contd

Section		Page
_0	STAGE SEPARATION	10-1
10.1	SUMMARY OF APPLICABLE BACKGROUND STUDIES	10-2
10.1.1	FR-1 Separation Linkage Description. . .	10-6
10.1.2	Preliminary Separation Trajectory Tradeoffs for FR-1.	10-8
10.1.3	FR-1 Study Summary	10-17
10.1.4	FR-1 Study Conclusions	10-19
10.2	FR-3 STAGE SEPARATION	10-20
10.2.1	FR-3 Stage Separation System Description	10-20
10.2.2	FR-3 Parameter Selection	10-20
10.2.3	FR-3 Study Conclusions	10-21
10.3	FR-4 STAGE SEPARATION	10-21
10.3.1	Pertinent Configuration Differences Between FR-4 and FR-1	10-21
10.3.2	FR-4 Study Conclusions	10-23
10.4	REFERENCES	10-23
11	MASS PROPERTIES	11-1
11.1	FR-3 CONFIGURATION	11-3
11.2	FR-4 CONFIGURATION	11-53

LIST OF FIGURES

Figure		Page
3-1	Battery Characteristics.	3-4
3-2	Solar Cell Performance at 0.4 V	3-6
3-3	Typical Fuel Cell Power Characteristics	3-9
3-4	Orbiter Electric Power System	3-12
3-5	Booster Electric Power System	3-14
3-6	Fuel Cell Reactant Supply System	3-16
3-7	Fuel Cell Water Removal System	3-17
3-8	Fuel Cell Thermal Control System	3-19
3-9	Three Kilowatt Fuel Cell Volt-Ampere Characteristic	3-20
3-10	Starting Time Period	3-21
3-11	Fuel Cell Equivalent Circuit	3-22
3-12	Fuel Cell Response	3-23
3-13	Main Bus and Feeders	3-26
3-14	Conventional Remote Bus Feed	3-26
3-15	Simple Remote Bus Feed	3-27
3-16	Electrical System Checkout Schematic	3-32
4-1	Typical Aerodynamic Flight Control System Schematic	4-5
5-1	Atmosphere Supply and Pressurization Control . . .	5-22
5-2	Atmosphere Purification Loop	5-25
5-3	Water Management	5-27
5-4	Waste Management	5-30
5-5	Personal Hygiene	5-32
5-6	Thermal Control System	5-33
5-7	Booster Thermal Control	5-37

LIST OF FIGURES, Contd

Figure		Page
6-1	FR-4 Booster Hydraulic Power Profile	6-2
6-2	FR-4 Orbiter Hydraulic Power Profile.	6-3
6-3	FR-3 Booster Hydraulic Power Profile	6-4
6-4	FR-3 Orbiter Hydraulic Power Profile	6-5
6-5	Actuation Subsystems, FR-4/FR-3 Booster	6-6
6-6	Actuation Subsystems, FR-4/FR-3 Orbiter	6-7
6-7	Typical Flight Control Actuation System (Scheme A)	6-11
6-8	Typical Flight Control Actuation (Scheme B).	6-12
7-1	Hydrazine Blend Fueled FR-1 APU Schematic . . .	7-3
7-2	Hydrazine Consumption as a Function of Output Shaft Horsepower	7-4
7-3	H_2-O_2 FR-1 APU Schematic	7-8
7-4	H_2-O_2 Consumption as a Function of Shaft Horsepower.	7-9
8-1	T-18 Vehicle Basic Configuration	8-3
8-2	Structural Arrangement.	8-21
8-3	Engine Compartment Structural Arrangment. . . .	8-25
8-4	Propellant Tank Structural Arrangement (Booster). .	8-29
8-5	Attachment of Centerbody to Propellant Tank . . .	8-35
8-6	Wing Support Structure	8-36
8-7	Payload Door - Hot Structure Arrangment	8-37
8-8	Side-Opening Payload Door Arrangement	8-40
8-9	Side-Opening Payload Door Configuration.	8-41
8-10	Payload Door Longeron Geometry	8-41
8-11	Payload Door Longeron Arrangement	8-42
8-12	Aft Body Structural Arrangement	8-44

LIST OF FIGURES, Contd

Figure		Page
8-13	Thrust Skirt Detail	8-45
8-14	Thrust Skirt Attachment Shear Splice	8-46
8-15	Forward Thrust Ring	8-47
8-16	Thrust Beam/Stabilizer Support Structure	8-49
8-17	Engine Gimbal Support Fitting	8-51
8-18	Stabilizer Center Section	8-52
8-19	Aft Body TPS and Trailing Edge Support Structure . .	8-54
8-20	Base TPS Support Structure	8-56
8-21	Wing Plan View, General Arrangement	8-58
8-22	Wing Cross-section	8-59
8-23	Flap Arrangement	8-60
8-24	Flap Expansion Joint.	8-61
8-25	Spoiler Arrangement.	8-62
8-26	Wing Pivot Fitting Arrangement	8-63
8-27	Stabilizer Plan View, Structural General Arrangement	8-65
8-28	Fin Cross Section.	8-66
8-29	Forward Attachment Point Arrangement	8-68
8-30	Aft Attachment Point Arrangement	8-71
8-31	Orbiter Heat Shield Materials and Design Temperatures for FR-3 and FR-4	8-73
8-32	Booster Heat Shield Design Temperatures and Materials for FR-3	8-74
8-33	Booster Heat Shield Design Temperatures and Materials for FR-4	8-75
8-34	Typical HS-188 Heat Shield Panel	8-76
8-35	Typical TD NiCr Heat-Shield Panel.	8-78

LIST OF FIGURES, Contd

Figure Page

8-36 Titanium Heat Shield (Flat Panel) 8-79
8-37 Titanium Heat Shield (Contoured Panel) 8-81
8-38 Typical Heat Shield Support Post. 8-82
8-39 Structural Arrangement, FR-3 Booster 8-83
8-40 FR-3 Booster, Hot Structure Concept 8-87
9-1 56-Degree Turnover Angle for FR-3 Booster . . . 9-5
9-2 FR-4 Booster and Orbiter Turnover Angle Comparison. 9-6
9-3 FR-3 Orbiter Turnover Angle Comparison 9-7
9-4 Landing Gear Arrangement 9-8
9-5 Landing Gear Arrangement 9-9
10-1 Separation Linkage Details. 10-7
10-2 Separation Linkage in Retracted Position 10-8
10-3 Separation Trajectories for Various Hinge-Release Angles 10-10
10-4 Separation Trajectories for Various Hinge-Release Angles with Thrust Decay 10-12
10-5 Separation Trajectories for Various Release Angles with Trimmer Stowed 10-13
10-6 Separation Trajectories for Various Release Angles with Nose Jet. 10-14
10-7 Separation Trajectories for Various Release Angles, Topside Booster 10-16
10-8 Link Load History 10-18
10-9 FR-4 Launch Configuration 10-22

LIST OF TABLES

Table		Page
3-1	Alternating Current Loads, 115/200 V 400 Hz, for Orbital and Booster Vehicles.	3-2
3-2	Direct Current 28 V Loads for Orbital and Booster Vehicles	3-3
3-3	Fuel Cell System Weights	3-28
3-4	Summary of Orbiter Electrical System Weights and Volumes	3-30
3-5	Booster Power System Weights	3-31
4-1	Design Parameters	4-1
5-1	EC/LSS Functions	5-2
5-2	Cabin Atmosphere Requirements.	5-16
5-3	Crew Water Balance	5-17
5-4	Mission Water Balance	5-17
5-5	EC/LSS Equipment Heat Loads	5-18
5-6	Orbiter Heat Load Summary	5-19
5-7	Booster Heat Load Reductions	5-20
5-8	Abbreviations and Chemical Symbols	5-21
5-9	Sensor Symbols	5-21
5-10	Component Symbols	5-21
5-11	Atmosphere Supply and Pressurization Control Redundancies	5-23
5-12	Atmosphere Supply and Pressurization Control Weights and Volumes	5-24
5-13	Atmosphere Purification Loop Redundancies	5-26
5-14	Atmosphere Purification Subsystem Weights and Volumes	5-26
5-15	Water Management Subsystem Backup and Emergency Modes	5-28

LIST OF TABLES, Contd

Table		Page
5-16	Water Management Weights and Volumes	5-28
5-17	Waste Management Backup Modes	5-29
5-18	Personal Hygiene Weights and Volumes	5-32
5-19	Thermal Control Redundancies and Backups	5-34
5-20	Thermal Control Weights and Volumes.	5-35
5-21	Orbiter EC/LSS Weight and Volume Summary . . .	5-36
5-22	Booster EC/LSS Weight and Volume Summary . . .	5-38
5-23	EC/LSS Expendables Rates	5-39
5-24	Weight and Volume Summary, 30-Day Mission . . .	5-40
6-1	Flight Control Acutation System Failure Analysis (Scheme A)	6-11
6-2	Flight Control Actuation System Failure Analysis (Scheme B)	6-12
6-3	Hydraulic System Weight (Pounds)	6-13
7-1	Technical Data - Hydrazine Blend Fueled Unit . . .	7-5
7-2	Technical Data - Hydrogen and Oxygen Fueled Unit	7-10
7-3	Summary of Total System Weight Estimates	7-12
8-1	Configuration Geometry Comparison	8-2
8-2	Properties of Heat Shield Materials.	8-10
8-3	Design Limit Load Factors	8-12
8-4	Factors of Safety	8-13
8-5	Pressure Differentials (Limit)	8-14
8-6	Minimum Gage Selection Guide for Various Materials	8-14
8-7	Material Selection Summary	8-16
8-8	Mechanical Properties of Candidate Materials . . .	8-17

LIST OF TABLES, Contd

Table		Page
8-9	Candidate Composite Materials Application	8-88
10-1	Nose Jet Parameters	10-15
10-2	Selected Parameters, FR-3	10-21
	FR-3 Mass Properties Data Compilation	11-3
	FR-4 Mass Properties Data Compilation	11-53

SECTION 1

INTRODUCTION

This report presents the results of subsystem studies and defines the significant characteristics of selected subsystems. The criteria to which subsystems must perform and the methods of analysis to permit selection are presented. Operational characteristics of the selected subsystems are provided to demonstrate compliance with overall vehicle system requirements. Both stowed-wing and fixed-wing configurations were studied. Since analysis began early in the study, some differences between the analyzed configurations and final configurations will be noted in some areas. These differences would not have affected concept definition but would have had a small effect on equipment sizing.

All subsystems are described with the exception of propulsion and electronics, which are in Volumes VI and VII respectively.

Commonality exists between the two- and three-element systems, booster to booster and orbiter to orbiter, primarily in the area of environmental control/life support and electrical power generation and distribution.

SECTION 2

GENERAL STUDY CRITERIA

Studies were directed toward the identification of systems applicable to a baseline seven-day orbital period. Both the orbiting elements and boost elements have fly-back turbojet engines. The orbiter engines are used primarily for powered landing and go-around capability. The booster engines are used to provide about an hour of cruise operation to return to base after a launch. All subsystems except propulsion and the vehicle thermo structure were designed to provide full operational capability after a single failure and to fail safe after a second failure. Selected concepts generally represent current or conservatively forecasted state-of-the-art technology to minimize development risk and cost.

The subsystems were selected and designed to meet the general mission characteristics listed below:

Phase	Time (min)	Altitude
Orbiter		
Ascent	6	0 to 50 n.mi.
Orbital	1 to 7 days	270 n.mi.
Reentry*	81	to 25,000 ft
Descent	10	25,000 to sea level
Approach and Land	5	sea level
Booster		
Ascent	2.8	to 170,000 ft
Reentry*	6.1	to 25,000 ft
Cruise	50	15,000 ft
Descent	15	15,000 to sea level
Approach and Land	5	sea level

---

*Wings deployed and airbreathing engines deployed and started between 25,000 and 15,000 ft.

SECTION 3

ELECTRICAL POWER GENERATION AND DISTRIBUTION

3.1 GENERAL VEHICLE REQUIREMENTS

Both the booster and orbiter vehicles must be supplied with electrical power continuously from liftoff through the landing phases. The orbiter has the more difficult requirement for power for a seven day period. During this period there are no mechanical sources from which electrical power may be converted. After reentry, turbojet engines become available as a source of power for the 15-minute aircraft mode.

For the booster, the period between liftoff and engine deployment is 8.8 minutes. The aircraft mode is a period of 70 minutes.

3.1.1 DESIGN REQUIREMENTS. The electrical system shall provide full operational capability, without transients or degradation of power quality, after the first failure. After the second failure, the system must provide sufficient power to maintain a safe condition. That is, equipment which is required for safety of return to base must be powered. The redundancy requirements apply to all elements of the system: generation, conversion, and distribution.

3.1.2 DESIGN OBJECTIVES. The operating life of the vehicle is ten years. Equipment used should be able to operate reliably during the use period and have minimum degradation during the dormant period between flights. High operational reliability requires that the system be assembled from a minimum number of major components each of which possess high reliability. The reusable aspect of the vehicle requires that selected major components should require minimum scheduled maintenance to minimize the annual cost of ownership. Unscheduled maintenance will be consistent with reliability achieved in operation.

3.2 LOAD ANALYSIS

The orbiter vehicle loads from boost to engine deployment are:

	Watts	
	Peak	Average
Guidance and Navigation	1143	908
Communication	300	205
Environmental Control/Life Support System (EC/LSS)	460	360
Lighting/Display	200	200
Multiplex Data	629	100
Two Compressors (H_2 and O_2)	6000	2000
Booster Engine Controls (per engine)	475	Negligible
Total	(Not additive)	3773

The booster vehicle loads from boost to engine deployment are:

	Watts	
	Peak	Average
Guidance and Navigation	873	638
Communication	300	205
EC/LSS	150	150
Lighting/Display	100	100
Multiplex Data	629	100
Booster Engine Controls (per engine)	475	120
Total	(Not additive)	1313

After engine deployment, the vehicle assumes the role of an aircraft, and additional loads not present during the boost and orbiting phases appear. These loads have been estimated based on aircraft experience and are preliminary in nature, but they offer the basis for making provisions on the vehicle to accommodate them.

The loads are presented in two sets. Table 3-1 gives those that are served best by a standard alternating current, 400 Hz system. Table 3-2 gives those usually supplied from standard direct current (28 V). These loads are assumed to be representative for both the orbital and booster vehicles.

Table 3-1. Alternating Current Loads, 115/200 V 400 Hz, for Orbital and Booster Vehicles

Fuel Boost Pumps (Three per Engine)	9,000 VA
Air Data, Turn and Bank, Speed Stability	400
Fuel Flow, Exhaust Gas Temperature Fire Detector	300
Ice Detectors	300
Windshield Anti-Ice	4,000
Scoop Anti-Ice (Various)	1,000
Fuel and Oil Quantity, Pressure Ratio	200
Anti-Collision Lights	500
Inboard/Outboard Landing Lights	4,000
Transformer/Rectifier Ac Load	3,500
Total	23,200 VA

Table 3-2. Direct Current 28 V Loads for Orbital and Booster Vehicles

Autopilot, Master Warning and Control	10A
Position Lights (External)	20
Duct Lip and Ice Detection	1
Valve Control (Various)	5
Landing Gear and Speed Brake Warning	10
Fuel and Oil Temperature Transducer	1
Emergency Pressurization	1
Anti-Ice Control and Indication	2
Flap Assymetric Control	2
Anti-Skid Control (Landing Load)	20
Rain Clearing, Overheat Control	1
Windshield Anti-Ice Control	1
Structural Overheat Warning	1
Total	75A

3.3 TRADE STUDIES

The trade studies reported in this section are primarily applicable to the orbiter since the booster electrical power requirement outside the atmosphere is small and system trades are not significant. Power sources studied included batteries, solar cells, and fuel cells. The parameters governing selection of the power source were principally weight and volume, although development status, reliability, and servicing requirements were among other parameters considered. Redundancy requirements will not be considered in these trades since only relative comparisons are desired. Further, the trade studies have been done based on an earlier load of 3660 watts, even though the present load analysis shows a load of 3773 watts. Again, only relative comparisons are desired.

3.3.1 BATTERIES. System electrical power can be supplied using either secondary or primary battery systems. Nickel-cadmium or silver-cadmium secondary batteries could be considered with silver-zinc considered the most likely primary battery candidate. Typical characteristics of these battery types are shown in Figure 3-1. The energy density of the silver-zinc system is significantly higher than either the secondary batteries, resulting in both a weight and volume advantage. Both the silver-zinc and silver-cadmium batteries have a much better wet-stand life than the Ni-Cd but lower rechargeability and overall life. For the vehicle operational life of 100 mission cycles over a 10 year period, it would appear that rechargeability is not a significant factor since battery capability and reusability would be limited by the life after initial activation. The Ni-Cd battery thus has limited reusability even though its rechargeability is quite good. Both the Ag-Zn and Ag-Cd batteries have similar wet-stand life and rechargeability. The higher energy density favors the Ag-Zn battery; therefore this battery was selected for the baseline comparison.

	ENERGY (W-HR/LB)			CAPACITY (AMP-HR/LB)			MID-POINT VOLTAGE		
DISCHARGE RATE→	5 hr	1 hr	15 min	5 hr	1 hr	15 min	5 hr	1 hr	15 min
BATTERY TYPE									
NICKLE-CADMIUM	15	12	10	12	10	10	1.25	1.20	1.00
SILVER-CADMIUM	25	20	16	22	19	18	1.20	1.10	0.80
SILVER-ZINC	53	45	39	35	31	30	1.50	1.50	1.30

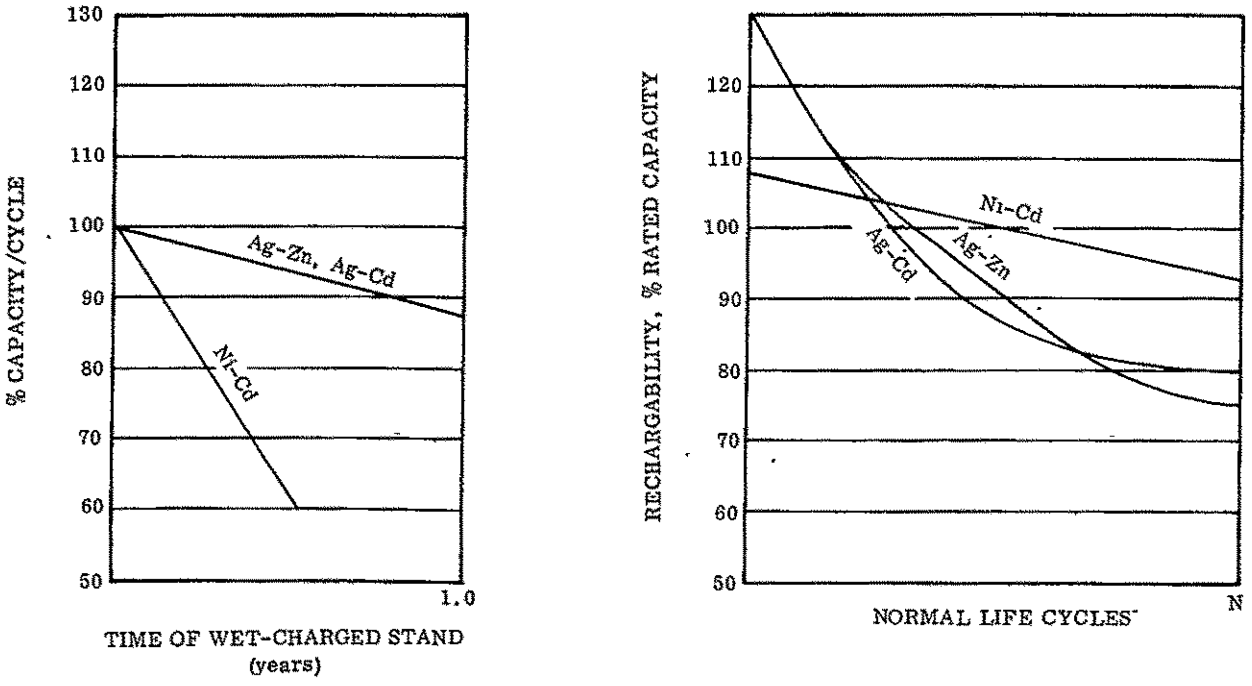


Figure 3-1. Battery Characteristics

Using an energy density of 53 watt-hr per pound and a nominal mission time of seven days (168 hours) the silver-zinc system was sized to provide an average power of 3660 watts. The system characteristics, allowing for 15% reserve capacity and accounting for increased packaging efficiency as battery size increases yield the following results.

Total Capacity, ampere-hr	25,200
Cell weight, lb	722
Number of cells	19
Installed weight, lb	15,050
Installed volume, ft^3	95.3

3.3.2 SOLAR CELLS. These systems have been successfully utilized in many space applications. The applicability is highly dependent on mission duration because installed weight and volume is determined almost solely by power level. The voltage developed by a silicon solar cell under full sunlight at maximum power conditions is about 0.4 volt. The orbiter load is essentially constant and if a sun tracking array is used, the array may be designed to operate near full power output. This full power output will be dependent on the array tracking accuracy, the condition of the cells, and the intensity of the solar illumination. The energy available from the array during sunlight operation will be a function of the thermal time constant of the array and the time during which the array is exposed to sunlight. This energy may be typically expressed by the integration of an equation having the general form of

$$E = \int_o^t K_1 + K_2 e^{-t/\tau}$$

where

E = energy

t = time in sunlgith

τ = array thermal time constant

K_1 & K_2 are constants dependent upon array physical characteristics

For a simplified trade study, however, consider only the solar cell maximum performance at a cell voltage of 0.4. A typical silicon solar cell performance characteristic is shown in Figure 3-2, which defines cell current as a function of solar intensity on the cell. The solar constant at the orbital altitudes of interest is 140 mw/cm^2. To approximate performance accounting for pointing error, thermal time constant, and radiation damage, use 85% of the solar constant or 119 mw/cm^2 for an estimate of array characteristics. From the figure, cell current = 26.1 ma/cm^2 giving a power of 10.5 mw/cm^2. Using a standard cell dimension of 2 cm × 3 cm then power = 62.7 mw/cell. The array must provide a power of 3660 watts continuously during the sunlit portion of the orbit and, in addition, must provide an equivalent amount to a storage

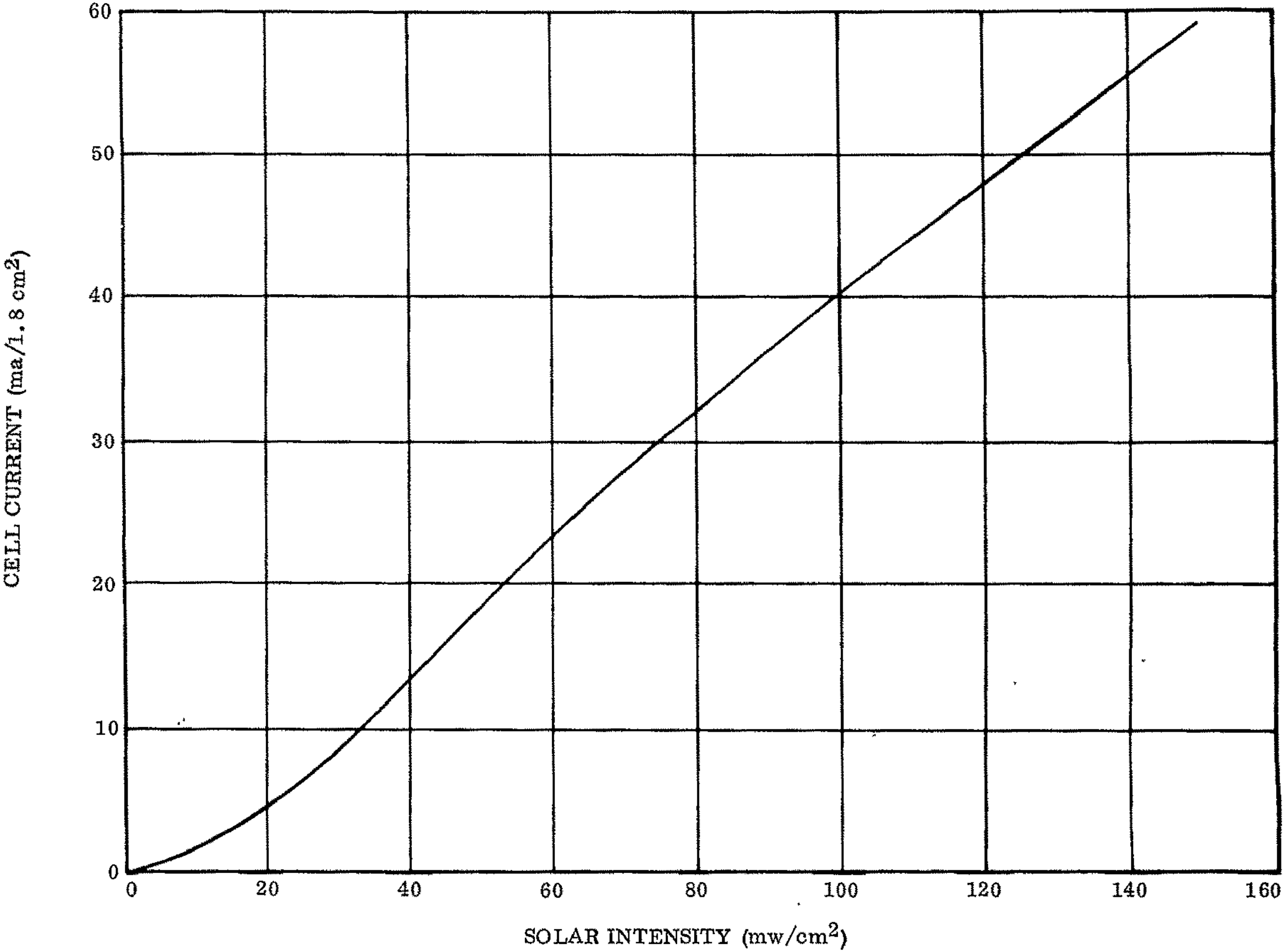


Figure 3-2. Solar Cell Performance at 0.4 V

system for use during the dark portion of the orbit (neglecting minimum/maximum, light/dark cycles). Thus the solar array must provide an average power input of 7320 watts per cycle.

Number of cells required = 116, 800
Array area = 1330 ft^2
Total area including 5% for added structure = 1400 ft^2

From previous studies of array construction, the array specific weight is on the order of 1 lb/ft^2 and the area to volume ratio is about 6 ft^2/ft^3

Array weight = 1400 lb
Array volume = 233 ft^3

A nickel-cadmium battery system designed for 25% depth of discharge per cycle was selected for the energy storage system. For a nominal 90-minute orbit, the energy per orbit to be contained by battery is 2680 watt-hours or 10, 700 watt-hours for 25% depth of discharge. For a 28-volt system and allowing 15% contingency capacity, a battery capacity of 440 ampere-hours is required. For a 1-hour discharge rate at 80° F, the nickel-cadmium battery specific capacity is 10 ampere-hours/lb. A summary of the battery system to supplement the solar cell system and total system characteristics, at 3660 watts average power, follows.

	Weight (lb)	Volume (ft^3)
Installed Battery	1162	8.1
Charger and Installation	28	
Solar Array	1400	233.0
Regulators, Wiring, Etc	100	2.0
Total	2690	243.1

In addition to the components shown above, the array will require a deployment and retraction mechanism and tracking system. The installation would add significantly to the weight and volume characteristics shown in the above table and would be relatively complex.

3.3.3 FUEL CELLS. The fuel cell offers a very high direct conversion efficiency for electrical power generation from electrochemical reactions. The H_2-O_2 fuel cell has a sound technological background and provides a very high specific energy. From electrochemical considerations the theoretical cell potential at 400° K is 1.16 volts with an electrical energy of 14.1 Kwh/lb H_2. Converting this to an equivalent 1 volt cell, the theoretical energy (E_t) is 12.15 E_a where E_a is the actual cell terminal voltage at the operational current density.

From the above, and combining H_2 and O_2 in stoichiometric proportion, this yields a theoretical reactant consumption rate ($\dot{w}$) of:

$$\dot{w} = 0.75\ Pt/E_a$$

where

P = Power level, kW

t = Operating time, hr

Because of parasitic loads such as coolant pumps, unit controls, instrumentation, and display and purging requirements, the actual reactant consumption rate will be on the order of $0.87\ P_t/E_a$. From this it is seen that minimum propellant consumption will be obtained when operating at the maximum cell terminal voltage. Typical fuel cell characteristics are shown in Figure 3-3 for end-of-life conditions.

This curve implies inherent voltage regulation in that a wide load range can be accommodated within a relatively narrow voltage band without requiring external regulation. This characteristic also provides good overload capacity with suitable design of the fuel cell thermal control system.

There are three major fuel cell developments currently in progress. Two of these (Allis-Chalmers and Pratt and Whitney) employ essentially the same technology, with the major difference being in the method of product water removal. Both approaches utilize KOH impregnated asbestos matrix contained between the electrodes of the cell. Allis-Chalmers uses a static water removal technique which employs a second KOH impregnated asbestos matrix spaced away from the H_2 electrode. Maintenance of a fixed KOH concentration difference between the matrices causes water vapor to diffuse from the fuel cell to the water removal matrix where it is then removed through a low pressure system maintained on the back side of the matrix. Pratt & Whitney causes water removal by carrying product water out of the cell in a recirculating H_2 stream. The P&W system results in a slightly higher parasitic loss, but eliminates the additional stack complexity and weight caused by including the water removal section. General Electric has followed a different technology in using a solid electrolyte composed of a sulfonic acid ion exchange membrane. Water removal is by wicking to a manifold where the water is drawn off. The A-C & P&W fuel cells currently exhibit a volt-ampere characteristic slightly better than the GE cell at start of life but both experience a significant performance deterioration with operational time whereas the GE membrane does not. Therefore, a design based on end-of-life performance would not appreciably favor any of the designs.

All of the fuel cells are currently considered to operate in the current density range of 100-200 amperes/ft^2, although much work is being done to explore operation at much higher current density (up to 4000 amperes/ft^2).

Present technology indicates that fuel cell systems can be built for a weight of about 35 lb/kW including all supporting accessories. Assuming that reactants are stored

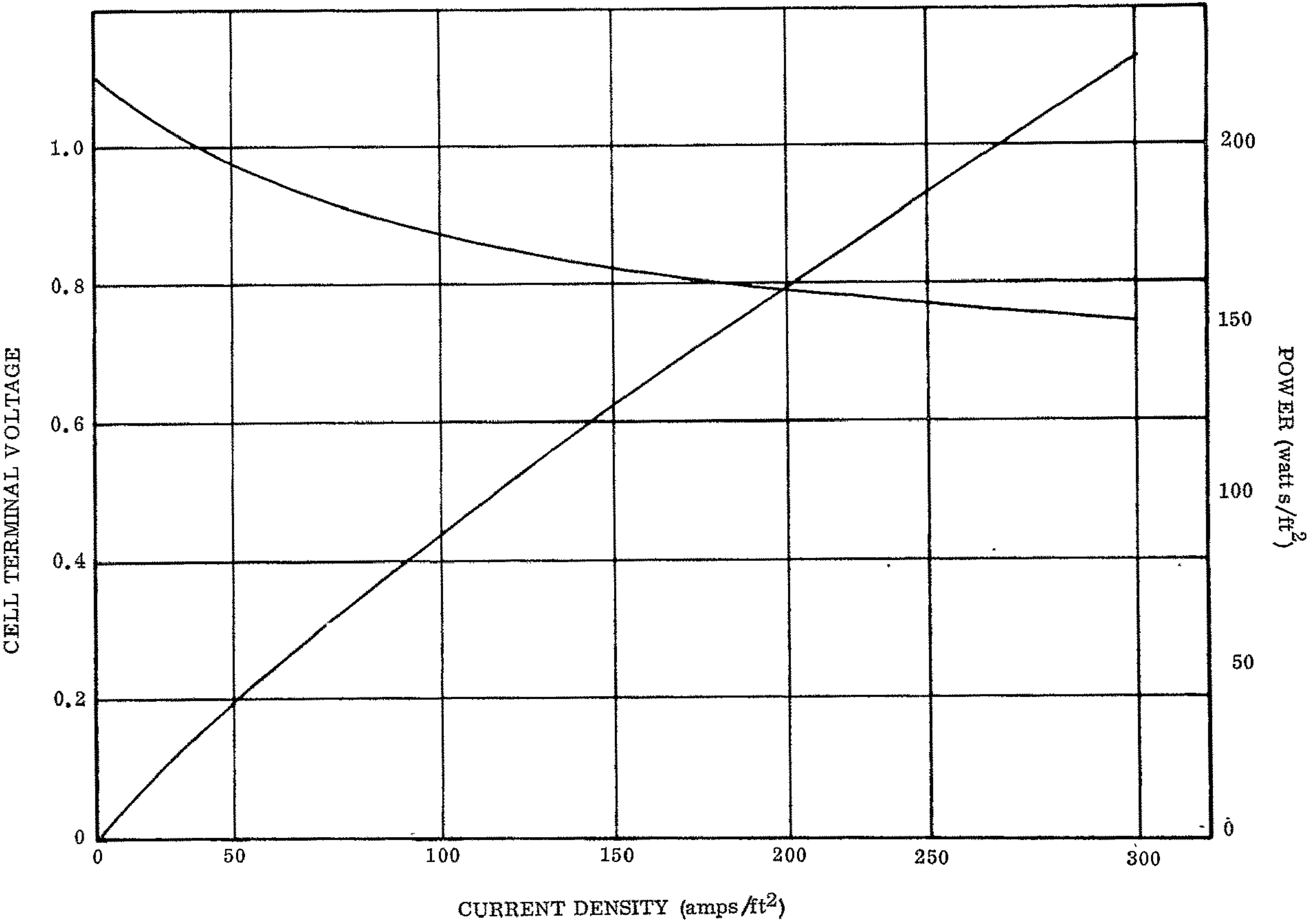


Figure 3-3. Typical Fuel Cell Power Characteristics

supercritically, a total fuel cell system weight can be estimated which accounts for reactants, reactant tankage and controls, and the fuel cell stack including accessories. Empirical expressions relating reactant tankage weight and volume to weight of usable fluid are:

For hydrogen

$$W_t = 13 + 1.78\ W_u,\ \text{lb}$$

$$V_t = 2.7 + 0.292\ W_u,\ \text{ft}^3$$

For oxygen

$$W_t = 17 + 0.205\ W_u,\ \text{lb}$$

$$V_t = 0.0228\ W_u + 0.5,\ \text{ft}^3$$

where

W_u = weight of usable fluid

(U) For a system to produce 3660 watts continuously for seven days (168 hours), and to meet failure criteria, it would be proposed to operate two fuel cells in parallel with an emergency battery to provide return power if both fuel cells should fail. Although normal operation is in a load sharing mode, each fuel cell and required reactants must be sized to provide full capacity for the entire mission. System characteristics of a single fuel cell system sized to the above criteria are summarized below.

	Weight (lb)	Volume (ft^3)
Oxygen	573	
Oxygen Tankage	135	12.7
Hydrogen	72	
Hydrogen Tankage	141	23.7
Fuel Cell	128	1.5
Lines, Valves, and Controls	8	
Total	1057	37.9

3.3.4 SYSTEM SELECTION. The extremely high weight and volume requirement of the battery system precludes serious consideration of this method of electrical power supply for missions of more than a few hours. Even for short missions, aircraft experience (F-106) has shown that the use of Ag-Zn batteries requires a disproportionate time for maintenance, servicing, and storage when not in use. Also, both the initial and recurring costs of such a system would be very high.

The use of solar cells will result in a large volume penalty although the weight of such a system does not appear to be unreasonable. There are, however, problems encountered because of the variable energy level provided by the array caused by thermal transients as well as variable illumination on the cells. The electrical power conditioning requirements are more severe than with either batteries or fuel cells and would require a greater quantity of more sophisticated equipment to meet reasonable using system demands. Recent developments in solar cell technology show great potential to produce flexible, thin-film CdS cells with conversion efficiencies nearly that of silicon cells. In addition to decreasing the problems of deployment, the cost of this type converter is much lower (approximately $50/watt) than that of silicon cells, which is approximately $400/watt. This system will bear further examination but was not selected for this study because of the problems mentioned and the apparent inherent advantages of the fuel cell.

The fuel cell is a very compact, highly efficient, well developed, direct conversion device. The inherent voltage regulation characteristic, simplicity of operation, and excellent overload capability are very desirable attributes and for nominal mission durations of a few days to a month would appear to offer the best method of electrical power generation for the orbital shuttle vehicles. This system was therefore selected as the baseline for this study.

3.4 POWER SYSTEM DESCRIPTION

3.4.1 ORBITER. The requirements for the orbiter vehicle are satisfied with the system elements shown in Figure 3-4. The operational sequence follows. The fuel cells are activated by opening the H_2 and O_2 valves using ground power. The fuel cell outputs rise in voltage and energize the bus. Ground power is removed and all vehicle loads associated with the boost, orbiting, and reentry may be energized as required. The fuel cells remain activated during the entire mission and share the system load. If either unit fails, the remaining unit continues to supply the entire load and the mission continues without affecting capability. If a second failure occurs, the battery is activated by dumping the stored KOH electrolyte into the cells. Power is now supplied only to those loads required for safety of return. Time limits imposed by limited energy storage require that the vehicle deorbit and reenter within a two-hour period. Stored electrolyte is used to provide long battery life in the inactivated state. Rarely will the battery be used, since it is an emergency source of energy. Once used, the battery is replaced as an expendable item.

The reentry period is considered to end upon engine deployment. After the engines are started, an engine-driven electrical power system is activated to accommodate those loads peculiar to the aircraft operating mode. The magnitude of this composite load suggest that the prime power be generated as alternating current. A variable speed-constant frequency (VSCF) system is shown. The oil-cooled, wide speed range, ac

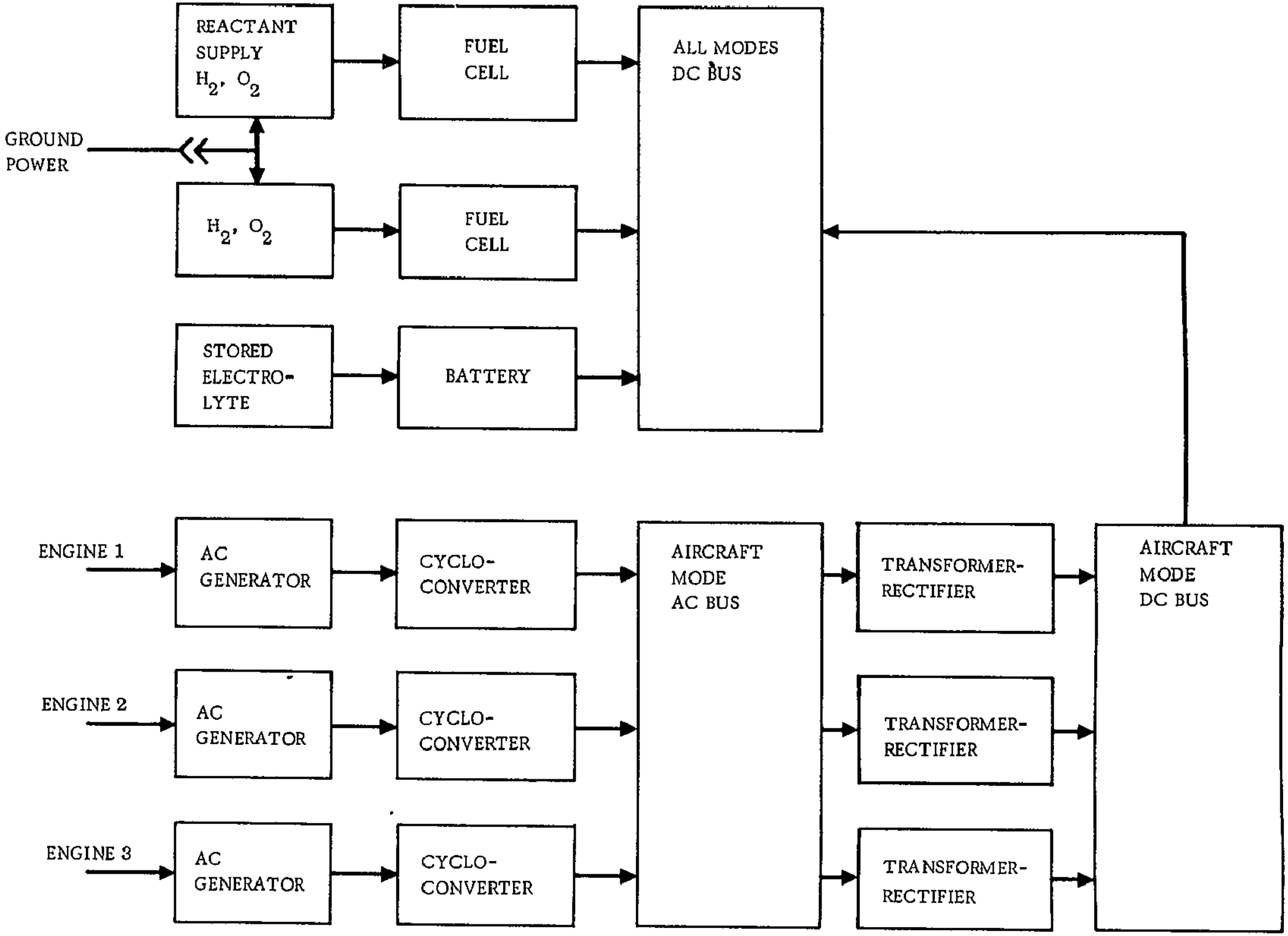

REACTANT SUPPLY H_2, O_2
FUEL CELL
ALL MODES DC BUS
GROUND POWER
H_2, O_2
FUEL CELL
STORED ELECTROLYTE
BATTERY
ENGINE 1
AC GENERATOR
CYCLO-CONVERTER
AIRCRAFT MODE AC BUS
TRANSFORMER-RECTIFIER
AIRCRAFT MODE DC BUS
ENGINE 2
AC GENERATOR
CYCLO-CONVERTER
TRANSFORMER-RECTIFIER
ENGINE 3
AC GENERATOR
CYCLO-CONVERTER
TRANSFORMER-RECTIFIER

Figure 3-4. Orbiter Electric Power System

generator is coupled directly to a high speed engine pad (10,000 to 20,000 rpm). The high output frequency of the generator (1200 to 2400 Hz) is converted to three phase, 115/200 volt, 400 Hz power by the cycloconverter. Direct current power is obtained simply by a transformer-rectifier. In addition to supplying aircraft mode loads, this system also supplies power to all of the loads on the fuel cell/battery bus in proportion to demand. All the dc sources can operate in parallel in any combination since their volt-ampere characteristics will be similar.

3.4.2 BOOSTER. A block diagram of the booster power system is given in Figure 3-5. The three batteries provide the redundancy needed. These are secondary batteries that are each capable of providing 10 minutes of power for the loads required between lift-off and engine deployment. The batteries are not recharged in flight, but rather are maintained in a charged state in the shop facility prior to installation.

After the engines are started for the aircraft mode, the provisions for electrical power are identical to the system described for the orbiter.

3.5 FUEL CELL THERMODYNAMIC CHARACTERISTICS

The H_2-O_2 fuel cell provides electrical energy directly through the overall reaction

$$2\,H_2 + O_2 \rightarrow 2H_2O$$

The anode reaction is

$$2\,H_2 \rightarrow 4\,H^+ + 4e^-$$

The cathode reaction is

$$4e^- + O_2 + 4H^+ \rightarrow 2H_2O$$

The total electrical energy available is obtained from complete conversion of all the free energy of the reaction. The maximum theoretical efficiency of the process is given by the change of free energy (ΔG) compared to the total enthalpy change (ΔH) of the reaction. For example:

$$\Delta G = \Delta H - T\Delta S$$

where $T\Delta S$ represents unavailable energy and an increase in system entropy. The efficiency is $100 \times \Delta G/\Delta H$. For standard conditions of one atmosphere and 25°C the maximum electrical energy available is 83% of the enthalpy change. Because of polarization losses, internal ohmic losses, and fuel cell parasitic losses, the present maximum practical conversion is more on the order of 65%, which is still very high compared to other electrical power producing processes.

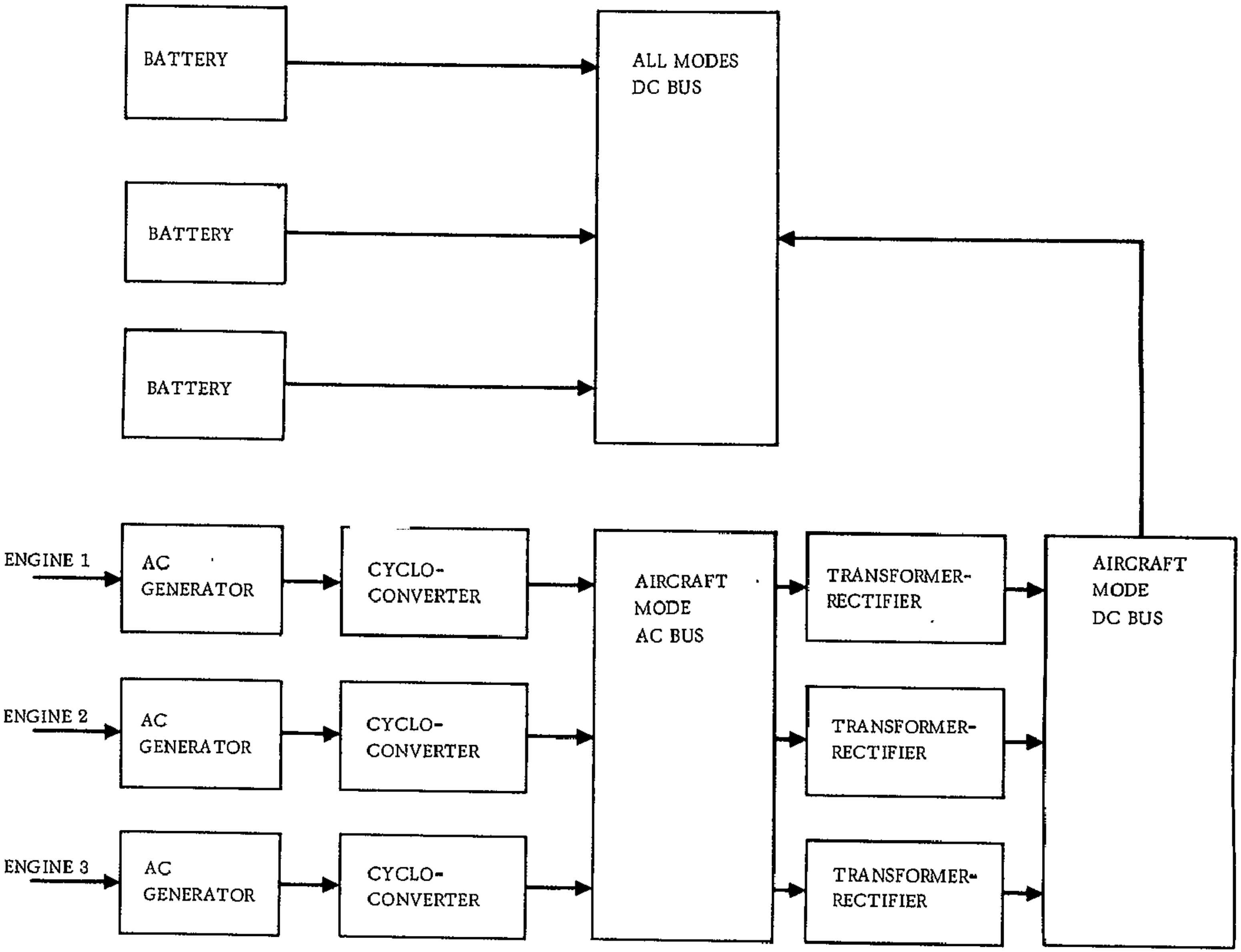

BATTERY
BATTERY
BATTERY
ALL MODES DC BUS
ENGINE 1
ENGINE 2
ENGINE 3
AC GENERATOR
AC GENERATOR
AC GENERATOR
CYCLO-CONVERTER
CYCLO-CONVERTER
CYCLO-CONVERTER
AIRCRAFT MODE AC BUS
TRANSFORMER-RECTIFIER
TRANSFORMER-RECTIFIER
TRANSFORMER-RECTIFIER
AIRCRAFT MODE DC BUS

Figure 3-5. Booster Electric Power System

There are essentially three distinct supporting systems necessary to permit proper fuel cell operation. These are: (1) a reactant supply system, (2) a product water removal system, and (3) a thermal control system.

3.5.1 REACTANT SUPPLY. In the present baseline system, it is proposed to use supercritically stored cryogens. The storage vessels must be provided with temperature control and pressure relief and regulating functions. To meet failure criteria it is proposed to use two fuel cell systems with completely redundant reactant supply. Provision to feed each fuel cell from either supply is included. A typical reactant supply system is shown in Figure 3-6. The system includes remotely activated isolation valves, heaters and supply pressure control. The purge valves shown are not necessarily part of the reactant supply system, but are necessary for periodically removing gaseous contaminants from the fuel cell. In operation the vessel pressure would be controlled to a predetermined value by use of the heaters. Operation of the fuel cell would be initiated by opening the shutoff valves and admitting reactant to the fuel cell. If a load is connected, the fuel cell may either bootstrap to operating temperature or it may be augmented by heat addition from an external source.

3.5.2 PRODUCT WATER REMOVAL. There are various methods presently used to remove product water from the reaction site. As previously explained, this may be accomplished by wicking, recirculation of excess reactant gas, or by differential pressure control between electrolyte impregnated matrices. Each system offers unique characteristics with resultant advantages and disadvantages.

If the product water is to be used in other systems (potable water, water boilers, etc.) the water vapor must be condensed and separated from the gas stream. An example of a water removal system is given in Figure 3-7. It is representative of the Pratt & Whitney system. This system recirculates excess H_2 through the fuel cell. Product water is swept from the cell in the H_2 stream and the gas mixture passed through a condensing heat exchanger where the water vapor is condensed. The resultant liquid is entrained in the H_2 stream and passed into a centrifugal liquid-gas separator. The separated liquid is then available for use in other systems. The water removal system will be common for both fuel cells with limited redundancy to meet failure criteria. If the water is to be used for human consumption, some treatment will be required to remove traces of electrolyte which occur to a small extent regardless of the configuration used. This is normally easily handled by chemical absorbents or an ion exchange column.

3.5.3 THERMAL CONTROL. The maximum percentage of ΔH which can be converted into electrical energy becomes smaller as temperature increases. On the other hand, internal ohmic losses decrease with an increase in temperature. At present the change in ohmic losses is more significant than the change in thermodynamic properties and it is advantageous to operate at high temperatures. The operational temperature is limited, however, by the thermal characteristics of the materials in the stack and principally by

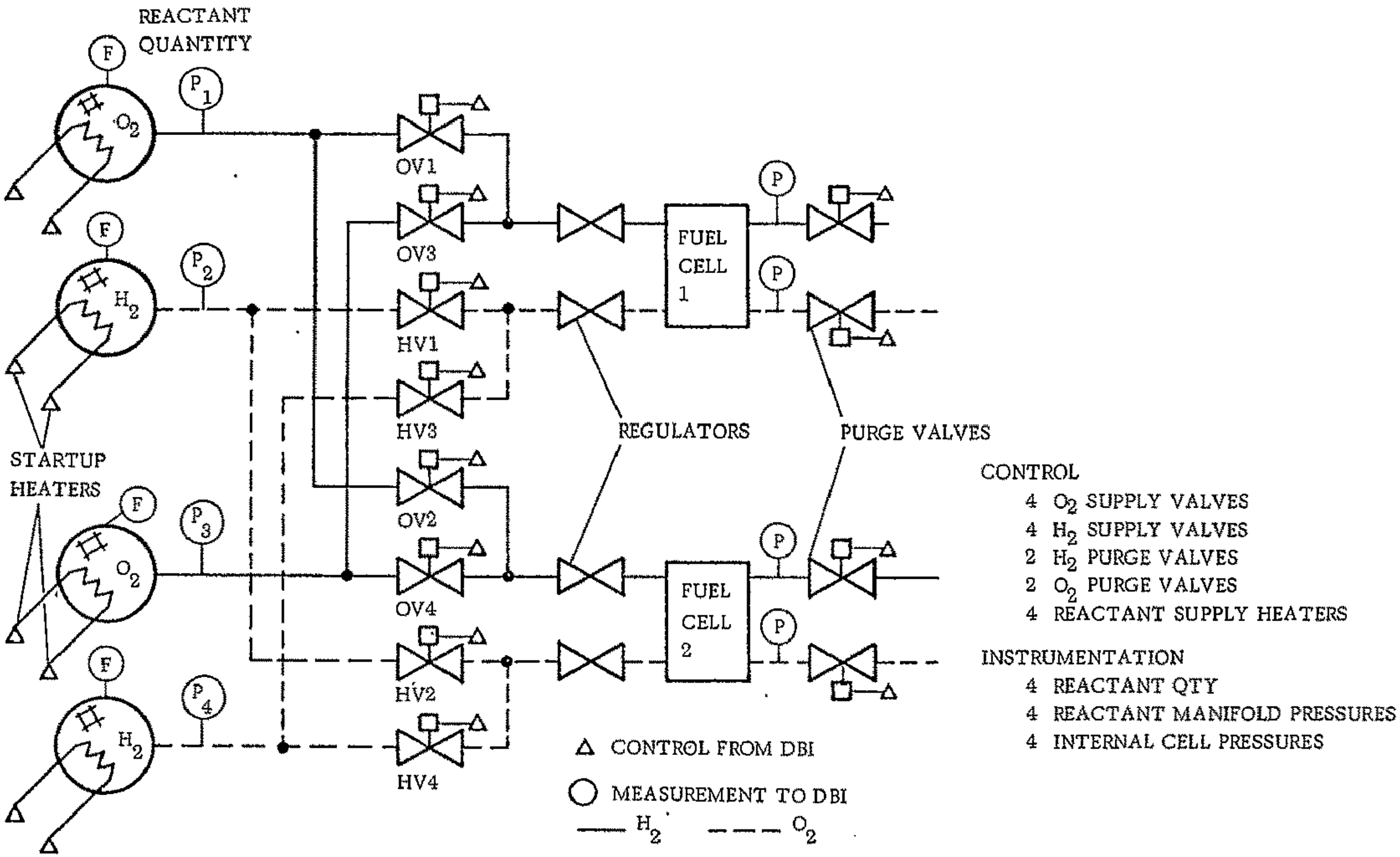
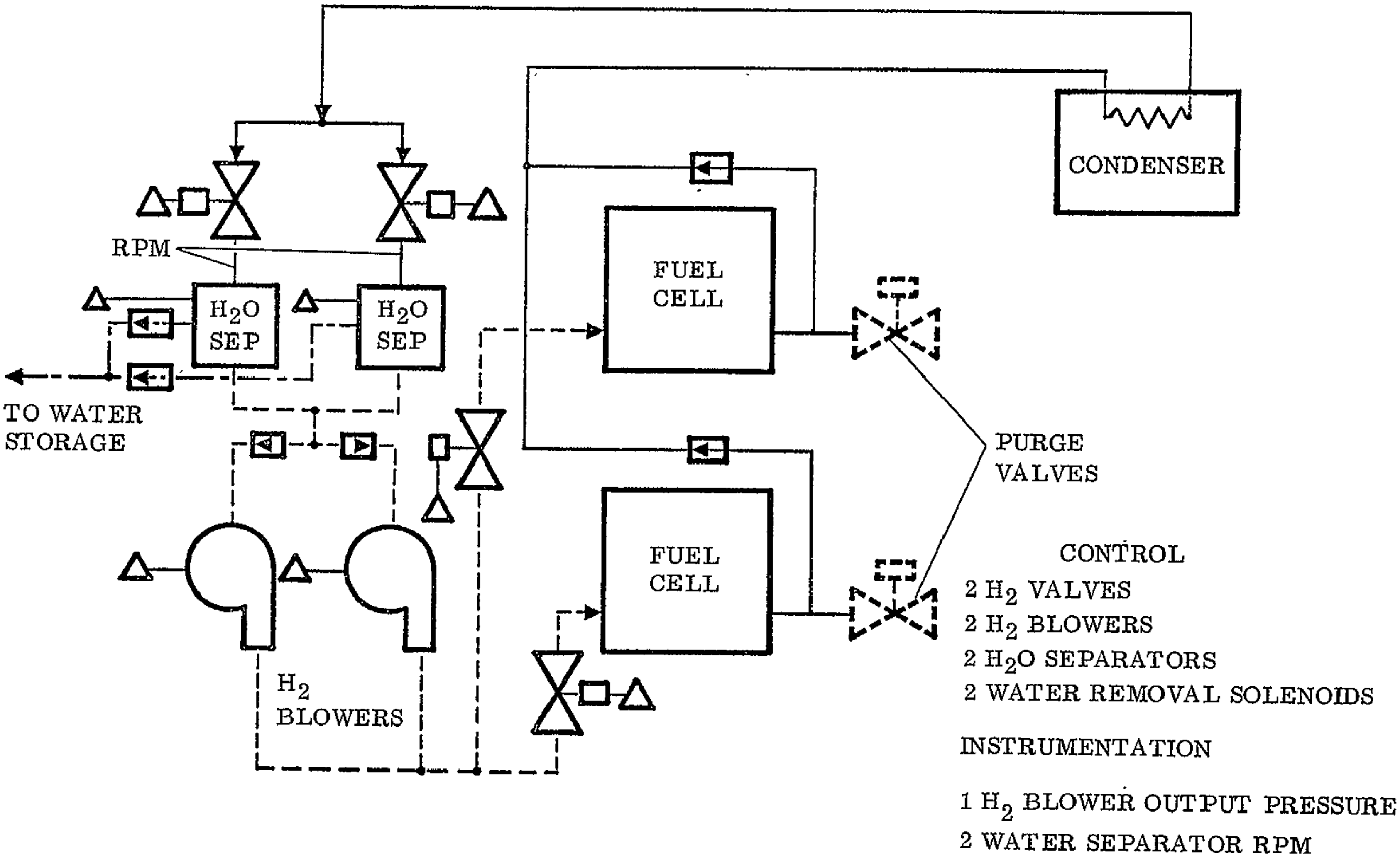

REACTANT QUANTITY
F
P_1
O_2
F
P_2
H_2
STARTUP HEATERS
F
P_3
O_2
F
P_4
H_2
OV1
OV3
HV1
HV3
OV2
OV4
HV2
HV4
FUEL CELL 1
FUEL CELL 2
P
P
P
P
REGULATORS
PURGE VALVES
CONTROL
4 O_2 SUPPLY VALVES
4 H_2 SUPPLY VALVES
2 H_2 PURGE VALVES
2 O_2 PURGE VALVES
4 REACTANT SUPPLY HEATERS
INSTRUMENTATION
4 REACTANT QTY
4 REACTANT MANIFOLD PRESSURES
4 INTERNAL CELL PRESSURES
△ CONTROL FROM DBI
○ MEASUREMENT TO DBI
—— H_2 – – – O_2

Figure 3-6. Fuel Cell Reactant Supply System


CONDENSER
FUEL CELL
FUEL CELL
RPM
H_2O SEP
H_2O SEP
TO WATER STORAGE
H_2 BLOWERS
PURGE VALVES

CONTROL

2 H_2 VALVES
2 H_2 BLOWERS
2 H_2O SEPARATORS
2 WATER REMOVAL SOLENOIDS

INSTRUMENTATION

1 H_2 BLOWER OUTPUT PRESSURE
2 WATER SEPARATOR RPM

Figure 3-7. Fuel Cell Water Removal System

the electrolyte impregnated membrane section. Operation at a too high temperature will degrade the volt-ampere characteristic and significantly shorten cell life.

For this reason, present technology limits the operating temperature to between 160°F and 200°F. The thermal control system must therefore be designed to control the stack internal temperature to this limit under the worst conditions of environment and stack load. All of the principal configurations now use liquid cooling with recirculation of the coolant through the fuel cell stack. An example of such a system is shown in Figure 3-8. This system supplies coolant to both fuel cells and is redundant with respect to pumps, filters, and control valves. The system shown interfaces with the vehicle main environmental control system. By proper heat exchanger design and control of fluid flow rate, no active thermal control is necessary on the condensing heat exchanger. Modulating control valves responding to coolant temperature leaving the stack will control stack temperature to the desired value. A startup heater is shown as an example of a method of heat addition to the stack to shorten startup time.

3.6 FUEL CELL ELECTRICAL CHARACTERISTICS

The characteristics of greatest importance are:

a. Steady state performance

b. Short circuit capability

c. Transient response, including startup, load application, and load removal

d. Parallel operation.

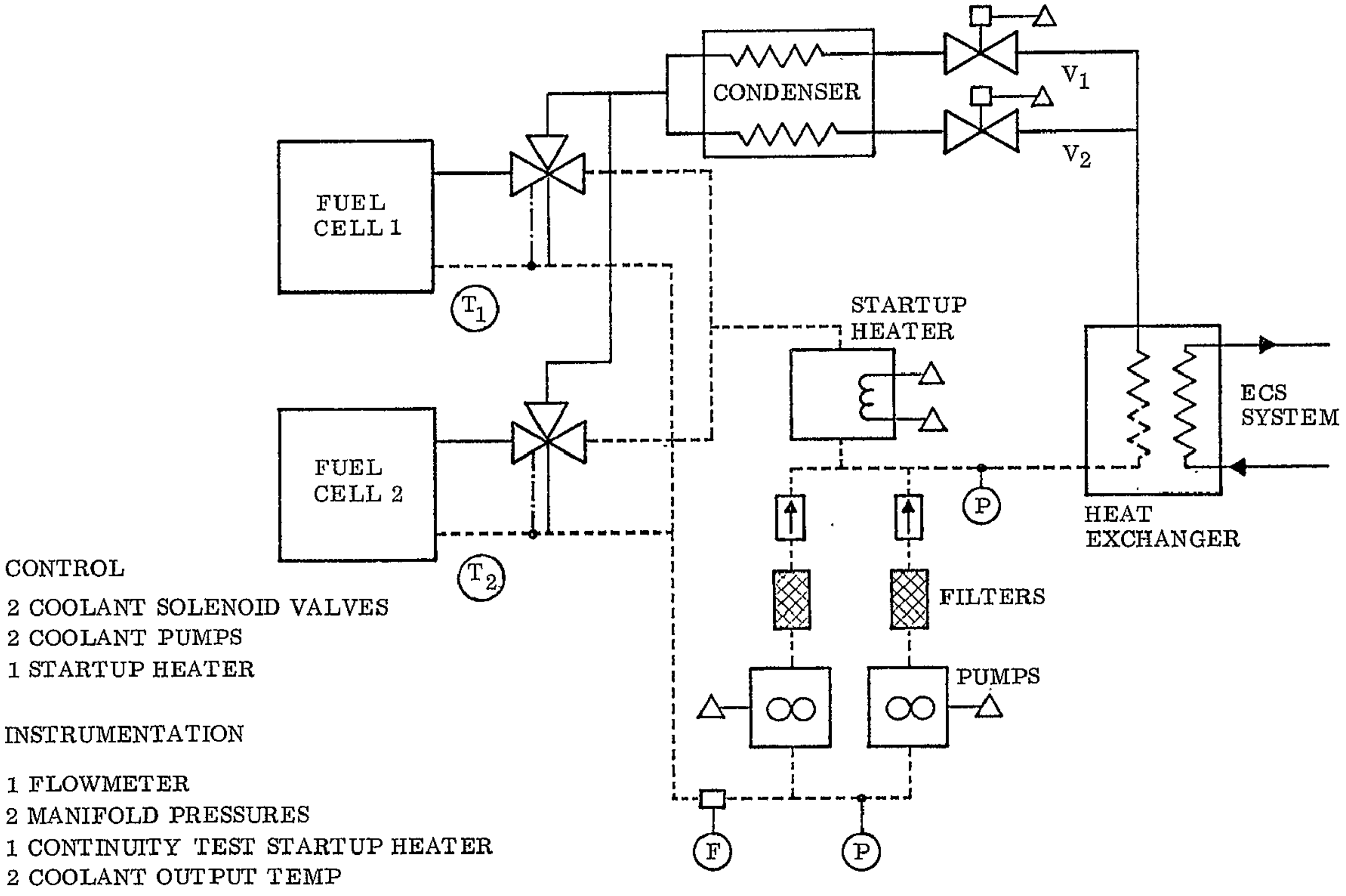
3.6.1 STEADY STATE PERFORMANCE. Current technology fuel cells exhibit a very flat volt-ampere characteristic as shown in Figure 3-9. This is the result of improved materials and higher operating temperature (180°F), reflecting a small internal resistance (approximately 0.05 ohm).

Low internal resistance, i.e., good inherent voltage regulation, is important and beneficial for several reasons:

a. Additional voltage regulating units are not required. This simplifies the system, thereby greatly increasing the reliability of the power source. Other beneficial results are reduced maintenance and cost of ownership.

b. The ohmic losses (I^2R) of the stack are reduced, resulting in improved efficiency at all loads.

c. Reduction in ripple voltage caused by cyclic loads, e.g., motor driven compressors or pulse type loads, minimizes system disturbances.


CONDENSER
V1
V2
FUEL CELL 1
T1
FUEL CELL 2
T2
STARTUP HEATER
ECS SYSTEM
HEAT EXCHANGER
P
FILTERS
PUMPS
F
P

CONTROL

2 COOLANT SOLENOID VALVES
2 COOLANT PUMPS
1 STARTUP HEATER

INSTRUMENTATION

1 FLOWMETER
2 MANIFOLD PRESSURES
1 CONTINUITY TEST STARTUP HEATER
2 COOLANT OUTPUT TEMP

Figure 3-8. Fuel Cell Thermal Control System

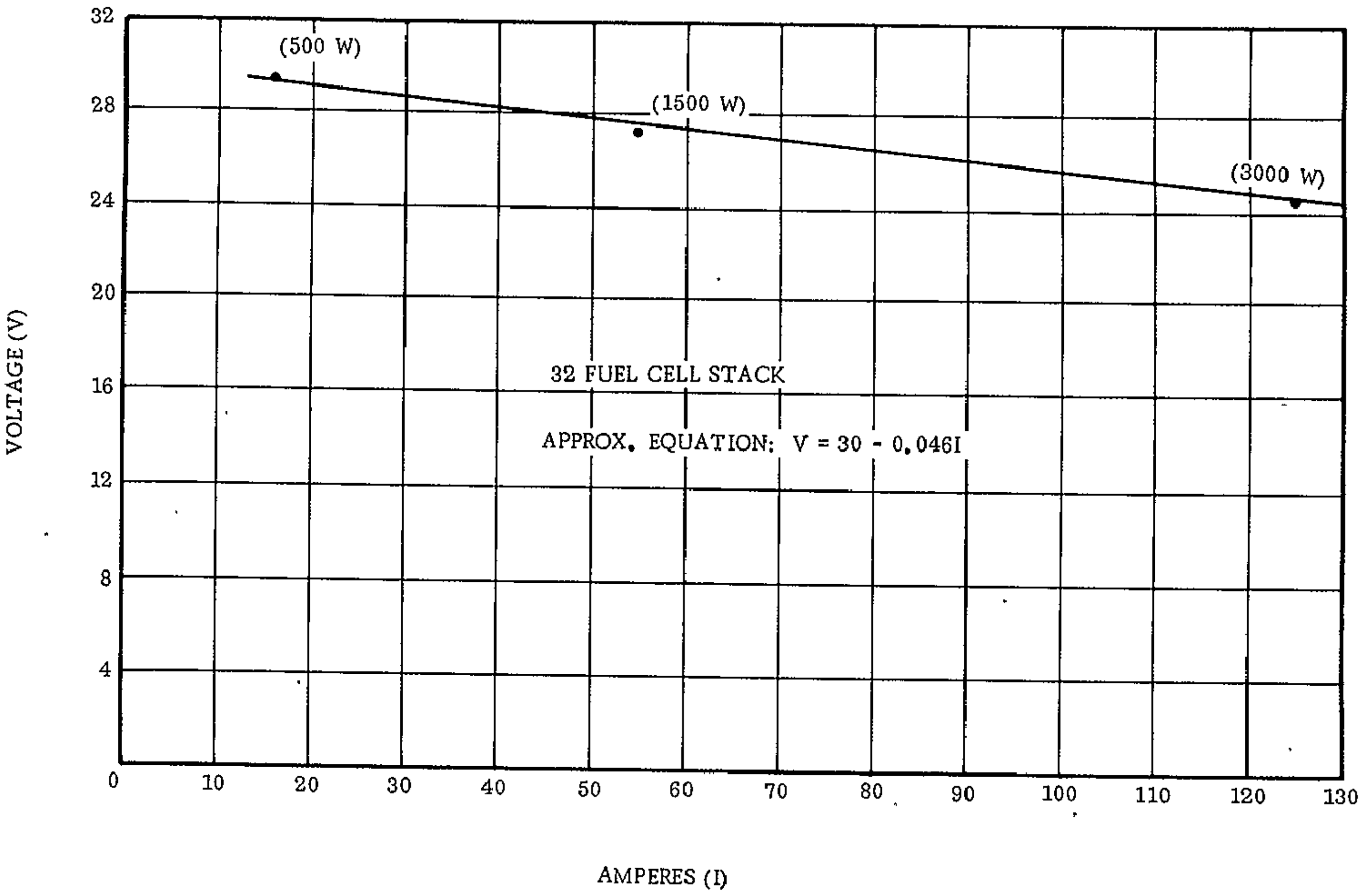


Figure 3-9. Three Kilowatt Fuel Cell Volt-Ampere Characteristic

d. Larger fault currents may be produced to burn clear feeder faults. Overloads and peak currents due to motor starting can be accommodated with less disturbance to the system voltage.

3.6.2 SHORT CIRCUIT CAPABILITY. It must be recognized that circuit protection devices (circuit breakers) must be rated to interrupt the fault current obtainable from the source. The design point is the fault current that may be produced when both stacks are operating in parallel. Each stack is capable of producing nearly 10 per unit current. Two 4.5 kVA units are therefore capable of producing nearly 3200 amperes. The actual value will be determined after cable impedances are known. Although existing electromechanical contactors and circuit breakers are able to interrupt fault currents in excess of this value and remain operational, solid state circuit breaker design has not yet reached this capability.

3.6.3 TRANSIENT RESPONSE

3.6.3.1 Starting Transient. The starting time period for two conditions after admitting H_2 and O_2 is shown in Figure 3-10. The upper curve shows the approximate shape of the open circuit (no-load) voltage curve versus time. If the output is loaded with a resistance R_L, the curve will be lower by the amount of voltage regulation inherent in the stack at some particular internal temperature T_1. As the stack temperature stabilizes, the output voltage will gradually rise to a higher voltage at temperature T_2. The time period to stabilization is dependent upon the stack losses.

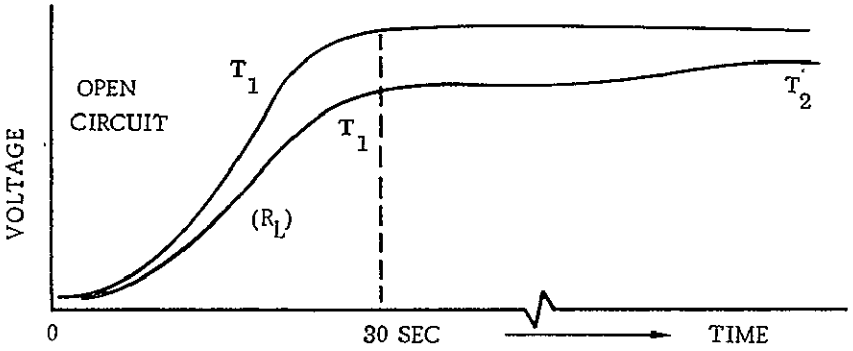


Figure 3-10. Starting Time Period

The startup period just discussed is representative of the General Electric fuel cell unit and closely approximates the performance of the Pratt & Whitney unit. In the case of the Allis-Chalmers unit, start up time is longer and more sensitive to application of load during the starting period. Start time has been reported to be in the range of one-half to one hour.

3.6.3.2 Load Application and Removal. The fuel cell responds immediately to changing current conditions caused by load switching. The shape of the voltage transient curve is best understood by examining the equivalent circuit of the fuel cell, shown in Figure 3-11.

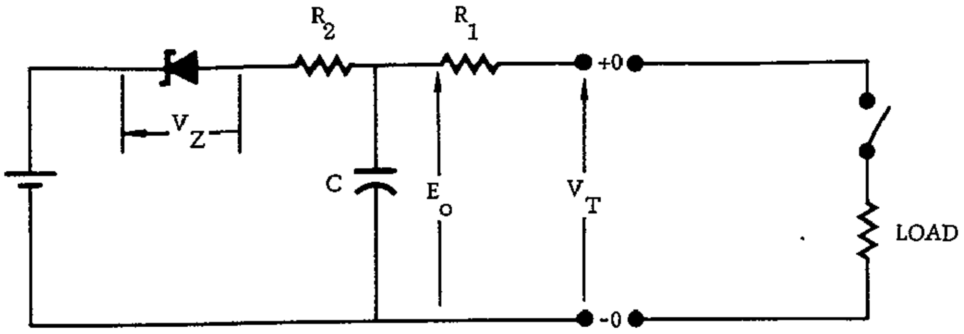


Figure 3-11. Fuel Cell Equivalent Circuit

This circuit is approximate, but is accurate enough for most needs. R_1 is linear and represents the ohmic losses of the stack. R_2 is non-linear to represent other than ohmic losses of the fuel cell. C is non-linear and represents the electrostatic energy storage and the storage of fuel and oxidant on the catalyst in the absorbed form. The Zener diode represents the threshold voltage which must be overcome at the reaction site (polarization). E_0 is an ideal voltage source.

Typical circuit constants for a 3 to 5 kW stack are approximately:

R_1	0.006 ohm
R_2	0.02 ohm
V_z	8.3 volts
C	2 farads
E_0	28.5 volts

A typical response of the fuel cell is shown in Figure 3-12. Upon application of load, an instantaneous drop in voltage occurs followed by an exponential decay, due to the large capacitance, to the final steady state value. The time constant is approximately 25 ms.

Upon load removal, the voltage rises abruptly, followed by an exponential rise in voltage to the steady state value.

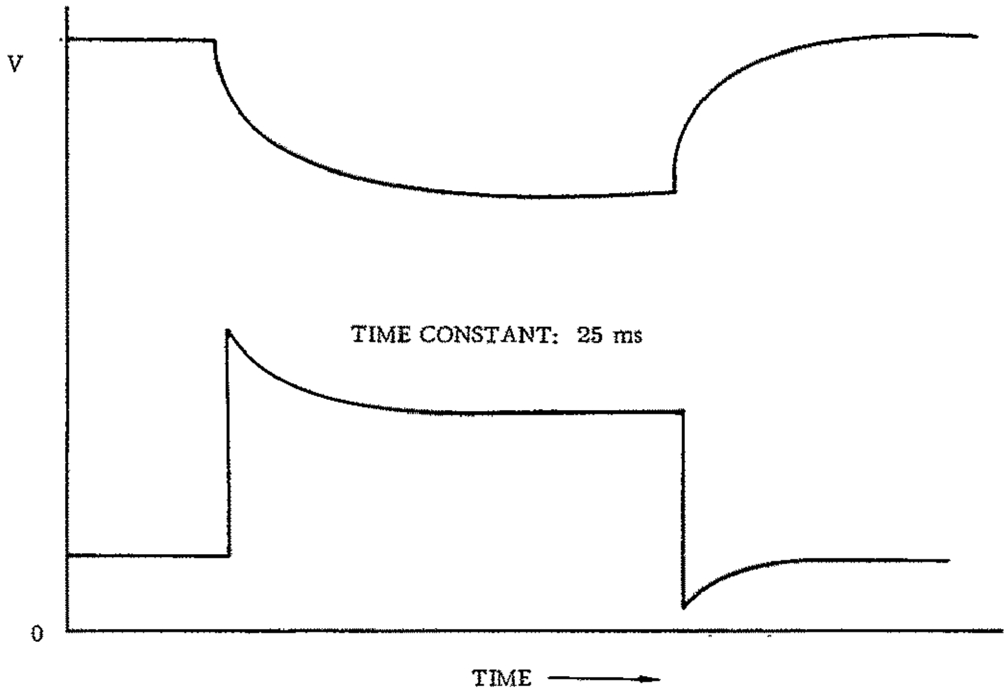


Figure 3-12. Fuel Cell Response

3.6.3.3 Parallel Operation. The slope of the volt-ampere curve of the stack is influenced very little by temperature and reactant pressure within the design operating range Similarly, the micro-volt per hour degradation can be minimized as a contributor to load sharing differential by purging at fixed time intervals. Variation in manufacturing tolerance will contribute some inaccuracy to load sharing.

Load sharing in this application is not critical, since each stack is designed to accommodate full system load plus some growth factor. Therefore, during parallel operation, each stack will be operating at half load or less.

To examine the influence of paralleling stacks with different internal resistance, the following analysis has been made:

a. A linear representation of the volt-ampere curve is sufficiently accurate. Assume each stack is rated at 100 amperes. Let the load resistance be 100 amperes at 28 volts. R Load = 28/100 = 0.28 ohm.

b. Assume stack A to have an internal resistance of 0.05 ohm and a linear intercept at no-load of 31 volts. Its characteristic equation is then

$$V_A = 31 - 0.05\, I_A$$

c. Assume stack B has an internal resistance 20% greater than stack A with the same no-load voltage; then

$$V_B = 31 - 0.06\, I_B$$

d. In parallel operation, their composite characteristic is

$$V_C = 31 - \frac{(0.05)\ (0.06)}{0.05 + 0.06}\,(I_C) = 3.1 - 0.0273\, I_C$$

e. The total current supplied to the load resistance of 0.28 ohm is

$$0.28\, I_C = 31 - 0.0273\, I_C$$

$$I_C = 101 \text{ amperes}$$

and the load voltage is: $(101)(0.28) = 28.28$ volts.

f. Stack A will contribute

$$I_A = \frac{31 - 28.28}{0.05} = 54.4 \text{ amperes}$$

and stack B will contribute

$$I_B = \frac{31 - 28.28}{0.06} = 45.33 \text{ amperes}$$

g. Load sharing is then

$$\frac{I_A - I_B}{I_A + I_B} = \frac{9.07}{99.73} = 0.09 \text{ p.u. } (9\%)$$

h. We may therefore conclude that a 20% change in internal stack resistance contributes less than 10% change in load sharing. A change in internal resistance of this magnitude may be expected as a result of gradual accumulation of gas impurities at the reaction site of the membrane material. This would be corrected at predetermined purge intervals.

3.6.4 TYPE OF POWER. The system described for the orbiting loads provides only one type of power: 28-volt direct current. No conversion to other voltages or frequency is provided, except in the aircraft mode. Each load will be designed to provide special voltages and frequencies as part of the unit.

The advantage of this method is:

a. The requirement for three inverters, sized for growth, is avoided. Inversion (dc to ac) is more complicated, less reliable, and heavier than conversion (ac to dc).

b. Inversion within each load unit, as needed, permits selection of the optimum frequency, voltage level, and regulation limits to suit each special requirement.

c. The additional complication of buses, ac circuit breakers, additional wire weight, and parallel operation is avoided.

During the aircraft mode of operation, it is more efficient to generate power in the form of alternating current. Further, relatively large loads, such as motor-driven pumps and heating loads are more efficiently served with alternating current at high voltage. Conversion to direct current (unregulated) using transformer-rectifiers is efficient, lightweight, and reliable.

Power quality for all sources should be required to meet the limits of MIL-STD-704. This specification should be used as the interface document between the power sources and the utilization equipment to eliminate the requirement for external voltage regulation and to capitalize on the inherent voltage regulation offered by the fuel cell.

3.6.5 BUSES AND FEEDERS. The fuel cells and other conversion units will be located in an unpressurized compartment aft of the pilot's compartment. The output of each fuel cell connects to the bus located in the pilot's compartment, which is near the center of electrical loads. These feeders are routed separately and are physically protected with insulating jackets against feeder faults. Each feeder enters the pressurized compartment with a separate bulkhead feedthrough.

The main bus is protected physically by an insulated enclosure and electrically by (1) diodes in each feeder to prevent feeder fault current contribution from more than the faulted source and (2) circuit breakers to each distribution circuit. This is shown in detail in Figure 3-13.

To satisfy the requirement for minimum wire weight, it may be necessary to establish sub-buses or remote buses near load centers other than the crew compartment. The distribution feeders between the two buses must also meet the design requirement for redundancy for failure. Failures may be either of two types: (1) open circuit (or high resistance connection), or (2) fault to ground. The latter requires protection so that a single fault cannot disable the complete system. The conventional method for providing a single feed between two points, without redundancy, is shown in Figure 3-14.

Each wire size and thermal protector is identical. If a feeder fault exists at point F in Figure 3-14, it is isolated by current contributions $I_1 + I_2$ and I_3 which are proportional to the fault resistance. Since the circuit protectors are equal rating, the protectors at both ends of the faulted feeder will open before the others since I_3 and $I_1 + I_2$ intersect the inverse time-current characteristic curve of the protector before I_1 or I_2. To provide the redundancy required for two failures one additional feeder is added, making a total of four wires. Each wire is rated to carry one-half the total load current. Selectivity is assured for faults on any two feeders.

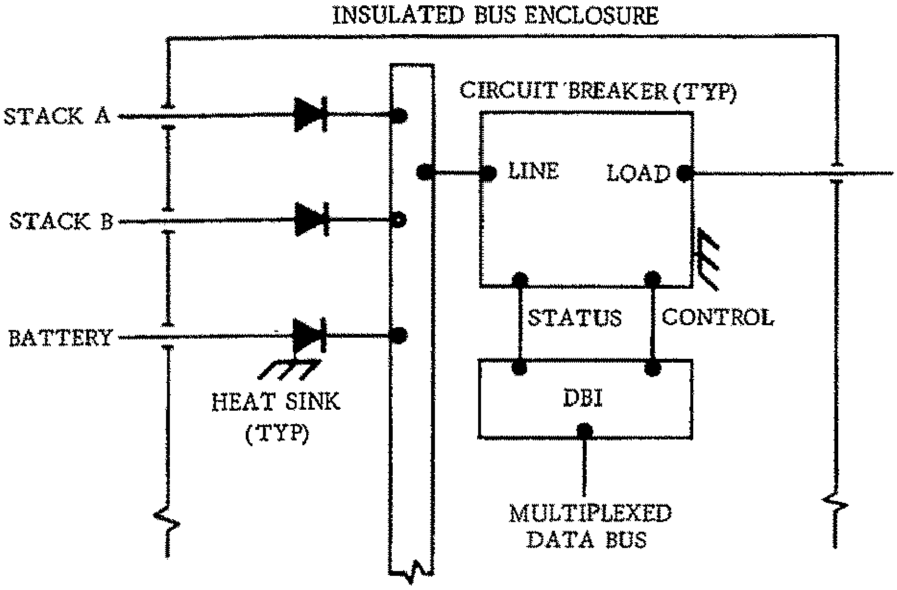

INSULATED BUS ENCLOSURE
STACK A
STACK B
BATTERY
HEAT SINK (TYP)
CIRCUIT BREAKER (TYP)
LINE
LOAD
STATUS
CONTROL
DBI
MULTIPLEXED DATA BUS

Figure 3-13. Main Bus and Feeders

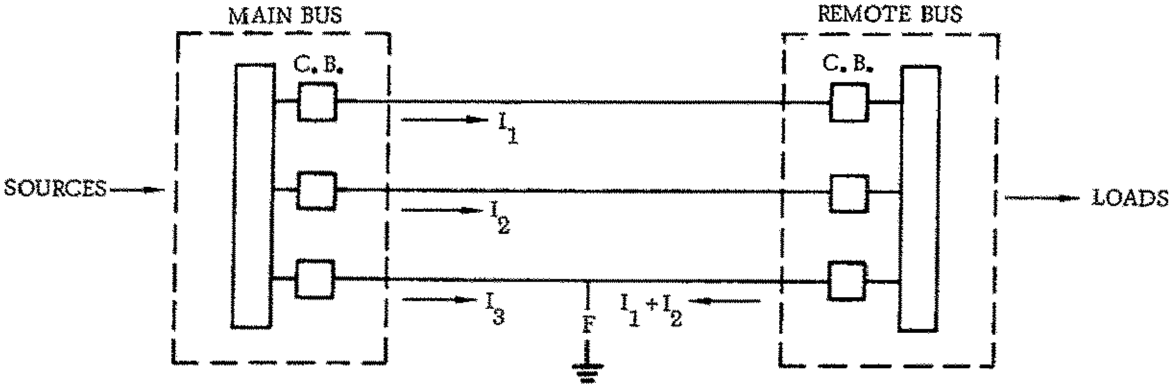

MAIN BUS
C. B.
SOURCES
I_1
I_2
I_3
F
$I_1 + I_2$
REMOTE BUS
C. B.
LOADS

Figure 3-14. Conventional Remote Bus Feed

A simpler and more reliable method for accomplishing the same result is shown in Figure 3-15.

A fault at point F is cleared and isolated by tripping the one affected protector since the remaining two feeders cannot contribute fault current. No marginal coordination of thermal protectors can exist. Each wire is rated for the full load current of the remote bus. This method for establishing a remote bus is preferred.

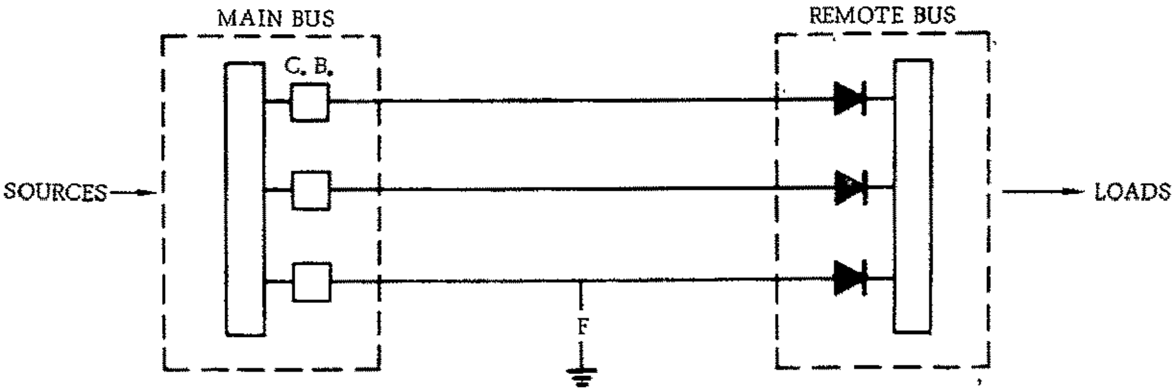


Figure 3-15. Simple Remote Bus Feed

3.6.6 DISTRIBUTION. The utilization equipment loads are supplied with power from the bus through remote controlled circuit breakers. The circuit breaker and wire connecting to the load are part of the subsystem or unit being served. The circuit breaker is the only switching element between the bus and the load. It is controlled (opened or closed) by a coded message supplied to the data bus interface (DBI) unit located within the bus enclosure. Each redundant load unit is provided with a separate circuit breaker and wire. This method is the most reliable for minimizing bus exposure and provides true load redundancy traceable to the source. The circuit breaker provides a status signal through the DBI link so that all commands may be verified. This method lends itself to central command and control for electric power management. The vehicle systems can be rapidly programmed for any mode of operation.

3.6.7 CIRCUIT BREAKERS. These units are located in the bus enclosure. They are solid state, using transistors or SCRs as the output load current switch. A voltage drop of about 0.4 to 0.5 volt is expected at full load and maximum ambient (base) temperature. Insulated heat sinking of each unit will be required.

Remote operation is accomplished through the DBI unit. Switching time will be typically fast; however, turn-on time may be programmed by a ramp function or by limiting the maximum inrush current. These features complicate the design by requiring additional solid state components.

The primary purpose of the circuit breaker is to afford thermal protection to the wire insulation so that smoke and fire hazards are avoided. The breaker is coordinated so that it will protect the wire in the bundle temperature environment. The breaker trips according to an inverse current-time curve. Indication of breaker position is provided for remote monitoring/display.

The development status of solid state remote-controlled circuit breakers is such that considerable effort must be expended if these units are to be available for the 1972 time period. Initial work has been done on remote-controlled electro-mechanical units which can be applied to the complete range of ratings. Solid state switching devices are becoming available, but have not been developed except for small ratings up to 10 amperes. The problems to be solved are: electromagnetic interference (EMI), heat sinking, fault current switching, DBI interface, minimum voltage drop, alternating current switching, and reliability demonstration.

3.7 ORBITER ELECTRICAL POWER SYSTEM WEIGHT AND VOLUME ESTIMATES

3.7.1 ORBITER

3.7.1.1 Fuel Cell System. The orbital system shown schematically in Figure 3-4 shows two fuel cells plus a remotely activated emergency battery. The system demand will be on the order of 3700 watts continuously, but to provide overload and growth margin, the fuel cell was sized to provide 4.5 kilowatts each. Normal operation is with both fuel cells sharing load, but the reactant supply must be adequate to accommodate single cell operation for the entire mission. Assuming the characteristic values used in the trade studies, the fuel cell system weights are given in Table 3-3.

Table 3-3. Fuel Cell System Weights

	Weight (lb)	Volume (ft^3)
Fuel cells: 157 lb × 2 =	314	2.5
Oxygen	1160	
O_2 Tanks	272	25.6
O_2 Residual @ 5%	57	
Hydrogen	144	
H_2 Tanks	274	47.4
H_2 Residual @ 5%	7	
Valves, Heater Lines	80	1.5
Total	2308	76.0

3.7.1.2 Isolating Rectifiers. Isolating rectifiers located at the central bus, including heat sink and mounting provisions, are 8 units weighing 1.0 lb each.

3.7.1.3 Dc Circuit Breakers. These will be solid state devices with remote reset and trip provisions operating from the digital bus interface. They provide status (on/off) to central computer for every load unit. Each load uses one circuit breaker as the only switching element to apply 28 Vdc power. In addition, the circuit breaker provices thermal overload protection for the wire run between the central bus and the load.

An estimated 120 loads at 0.6 pound per circuit breaker totals 72 pounds. This weight includes the mounting and heat sink provisions for each unit.

3.7.1.4 Wiring, Fuel Cells to Bus. The power sources are assumed to be mounted in the compartment aft of the pilot's compartment. The bus is located at the aft bulkhead of the pilot's compartment within the pressurized area.

Each wire runs 15 ft. Three harnesses, each containing three #8 gauge wires, are routed separately.

8 gauge = 63 lb/1000 ft

9 x 15 = 135 ft	8.5 lb
Physical protection, each harness (0.5 lb)	1.5
Harness supports, feedthroughs, connectors	2.0
	12.0 lb

3.7.1.5 Silver-Zinc Battery. This battery has stored electrolyte, squib actuated, for failsafe, degraded mode of operation. This unit is quiescent until two fuel cells have failed.

	Weight (lb)	Volume (ft^3)
Assume 2 kW load for 2.5 hours	120	1.3

3.7.1.6 Ac Generators. Alternating current loads plus growth provisions size the generator at 40 kVA for the aircraft mode of operation. A direct engine-drive ac generator is used, using a cycloconverter principle to construct three-phase 400 Hz 115/200 V at 40 kVA

	Weight (lb)	Volume (ft^3)
Generator: 50 lb x 3 engines	150	1.65
Converter: 50 lb x 3	150	4.35
Oil cooler, oil, lines: 10 lb x 3	30	-
	330	6.0

3.7.1.7 Converter. Direct current conversion (transformer-rectifier) from 400 Hz for the aircraft mode loads plus a growth provision requires 200-ampere 28 Vdc units.

Three units weighing 20 lb each totals 60 lb.

3.7.1.8 Wiring, Generators to Bus. This wire is for generator feeders and transformer-rectifier feeders.

Generator: 40 ft per feeder x 0.5 lb/ft x channels	60.0 lb
Transformer-Rectifier: Same as fuel cells to battery	8.5
Physical protection (all feeders)	5.5
Supports, feedthroughs, connectors	10.0
	84.0 lb

Insulated Bus Enclosure for Ac and Dc Buses	20 lb
Ac Circuit Breakers: Estimated 70 required at 0.6 lb each	42 lb

3.7.1.9 Summary. The orbiter electrical system weights and volumes are summarized in Table 3-4.

Table 3-4. Summary of Orbiter Electrical System Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
1. Fuel cell system	2308	76.0
2. Isolating rectifiers	8	-
3. Dc Circuit breakers	72	0.3
4. Wiring, fuel cells to bus	12	-
5. Silver-zinc battery	120	1.3
6. Ac generators	330	6.0
7. Converter	60	1.5
8. Wiring, generators to bus	84	-
9. Bus enclosure	20	2.0
10. Ac circuit breakers	42	0.2
Total	3046	86.8

Note: Items 2, 3, and 10 are contained within item 9 volume.

3.7.2 BOOSTER POWER SYSTEM WEIGHTS. The booster electrical power requirements are essentially the same as for the orbiter with the exception of the orbital time requirements. The primary difference is in the substitution of Ni-Cd batteries in place of fuel cells. A summary of the booster electrical power system weights is given in Table 3-5.

Table 3-5. Booster Power System Weights

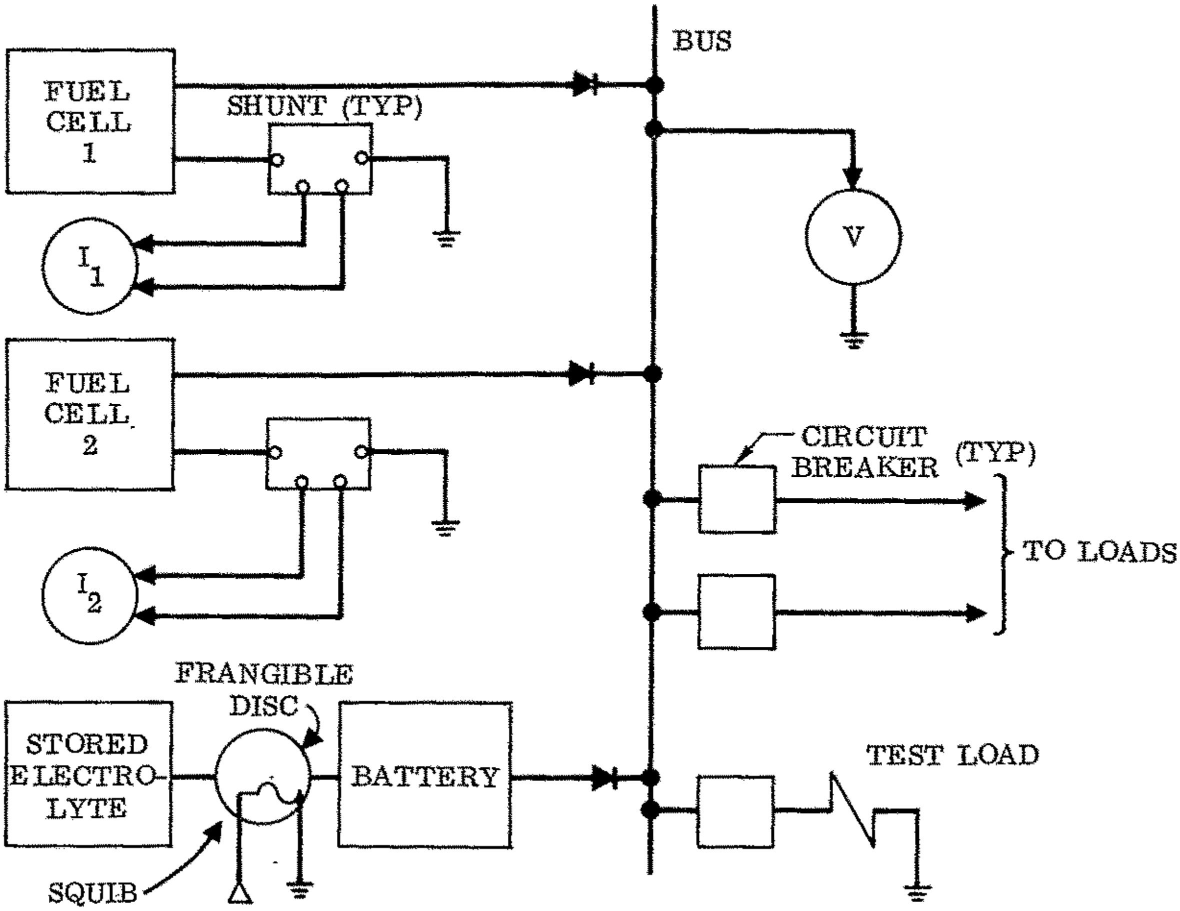
Item	Weight (lb)
Ni-Cd Batteries	50
Dc Circuit Breakers	72
Wiring, Battery to Bus	4
Isolation Rectifiers	4
Ac Generators	150
Cyclo-Converters	150
Oil Cooler	30
Transformer-Rectifier Units	60
Wiring, Generators and TRs to Bus	84
Bus Enclosure	20
Ac Circuit Breakers	42
Total	666

3.8 FAILURE MODES, DETECTION AND ACTION

A brief examination was conducted to study the failure modes, ability to sense failures, and to provide fault isolation for purposes of onboard checkout and servicing. A detailed discussion of the interfacing and operation of the electronic checkout system is presented in Volume VII, Integrated Electronics.

3.8.1 ELECTRICAL SYSTEM. Measurements conducted on the output of the fuel cells cannot be made until they have reached normal operating temperature (160 to 180°F). This is because of the change in the slope of the volt-ampere curve with temperature; i.e., at low temperature (say 70°F) the slope is relatively steep, resulting in poor inherent voltage regulation. Within the normal operating temperature range, however, the slope is quite flat. The no-load to full-load voltage will be approximately 31 to 27 volts at the main bus. A thermal control system maintains the correct temperature range.

Checkout of a typical system (Figure 3-16) is accomplished by measurement of each fuel cell current and bus voltage. Voltage is measured to establish that the minimum acceptable bus voltage is available for any given load condition. The current output of each unit is measured to assure that load sharing is within acceptable limits. An appropriate test load will be used to achieve a level of current per fuel cell that will provide confidence that the fuel cells and their associated feeders can transmit the required power.


BUS
FUEL CELL 1
SHUNT (TYP)
I_1
V
FUEL CELL 2
I_2
CIRCUIT BREAKER (TYP)
TO LOADS
FRANGIBLE DISC
STORED ELECTRO-LYTE
BATTERY
TEST LOAD
SQUIB

INSTRUMENTATION

2 CURRENT SHUNTS
1 VOLTAGE
1 RESISTANCE MEASUREMENT (SQUIB)

CONTROL

1 TEST LOAD

Figure 3-16. Electrical System Checkout Schematic

The mathematical equation of each fuel cell and the composite equation of the paralleled fuel cells may be written from test data. These mathematical models are stored in the computer and fuel cell performance is measured against the standard. The general equation of a fuel cell may be approximated by a linear representation, sufficiently accurate for all purposes, as follows:

$$E - IR = V$$

where E is the open circuit voltage, taken as the straight line intercept, I is the load current, R is the apparent internal resistance of the fuel cell plus the resistance of connections, wire, and isolating rectifiers between the fuel cell and the bus, and V is the resulting terminal voltage at the bus.

3.8.2 REACTANT SUPPLY. A diagram of this system is given in Figure 3-6.

3.8.2.1 Heater Failure. A heater failure will be noted by the pressure measurement in the storage vessel outlet line. Failure to observe an increase in pressure when the heater is activated will require isolation of the affected vessel and changeover to the redundant vessel.

3.8.2.2 Line or Vessel Leakage. If the leakage is significant, it will be detected by excessive activation of the heaters and by more than normal rate of decrease in reactant quantity. This will require isolation of the affected vessel and changeover to the redundant vessel. Leakage between the vessel isolation valves and the fuel cell module will be detected by charging the fuel cell to regulated pressure and then closing the isolation valves and observing the pressure decay. Leakage may or may not be sufficient to warrant changeover to the redundant system. Operational standards can be defined to regulate the appropriate response to a leakage indication.

3.8.2.3 Purge Valve Malfunction. Following the leakage check above, opening and closing the purge valves and observing the step change in pressure will verify normal valve operation. Action in the event of purge valve malfunction depends on the malfunction. If the valve will not open, operation can be continued and fuel cell performance will degrade as a function of the impurities collected in the cells. Eventual changeover to the redundant module will be required when module load sharing becomes out of tolerance. If the valve will not close, the fuel cell module will be isolated and operation continued on the second module.

3.8.2.4 Regulators. A regulator malfunction will be indicated by inability to maintain fuel cell pressure, by maintenance of improper fuel cell pressure, or, by inability to pressurize the module. Operation may be continued if fuel cell pressure is off specification but within a limited (possibly emergency mode) band. Otherwise operation will be continued with only the second module.

3.8.2.5 Fuel Cell Module. Cell integrity and membrane permeability will be verified by the leakage check of Section 3.8.2.2 if no leakage is observed.

3.8.3 PRODUCT WATER REMOVAL. A diagram is given in Figure 3-7. Failure of any of the components that cause water not to be removed from the cells will cause an immediate degradation in electrical output. This will be detected by load-sharing being out of tolerance and a drop in module voltage.

3.8.3.1 Water Separator. Each water separator can be started in turn and operation verified by observing that rpm rises to proper value. Water separator rpm is also sensed to verify continued operation. Failure of the separator to pump water will be reflected in a decrease or cycling in separator rotational speed. The faulty separator is then isolated and the stand-by separator brought into operation.

3.8.3.2 Water Separator Isolation Valves. With the recirculation loop activated, each valve can be opened and closed in a sequence which causes the operation to be detected by a change in blower output pressure. Either valve can be detected, if faulty.

3.8.3.3 Blowers. Each blower can be checked by observing blower discharge pressure after starting. Continued monitoring of this pressure will detect incipient or actual failure. If failure occurs, the blower is turned off and the second blower started. Check valves ahead of each blower prevent short-circuiting.

3.8.3.4 Module Isolation Valves. With the recirculation system operating, these valves may be sequentially operated and the response detected by change in blower discharge pressure. A failed-closed condition of either valve will require operation on the remaining module. A failed-open condition will permit continued operation even if either fuel cell required shut-down for other reasons.

3.8.3.5 Condenser. Condenser failure will be detected as an inability to remove product water from the cells. Voltage of both modules will decrease. Load sharing may or may not go out of tolerance.

3.8.4 THERMAL CONTROL. See Figure 3-8 for a diagram of this system.

3.8.4.1 Liquid Pumps. Operation of either pump is verified by sequentially activating each pump and observing the resultant coolant flow. A redundant pump is activated in the event of pump failure. Check valves at the pump discharge prevent fluid from short-circuiting around the defective pump. System blockage may also be observed by monitoring pump inlet and discharge pressures.

3.8.4.2 Filters. Blocked or loaded filters will be detected by activating each pump in turn and observing pump inlet and outlet pressure and also by a low flow indication.

This may also indicate stuck or blocked check valves. Operation may be continued if necessary until a further fault is observed either in inadequate control of module temperature or lack of water removal.

3.8.4.3 Startup Heater. The heater may be tested for continuity and warm up time observed to determine that the heater is operational. The use of the heater is optional because the module is capable of boot-strapping to operational temperature.

3.8.4.4 Thermal Control Valves. Improper operation of the thermal control valves will be evidenced by inability to maintain a pre-selected coolant discharge temperature from the module. This may be the result of a faulty temperature sensor and/or control or could be caused by blockage in the valve. The latter fault would be detected as a low flow indication in the coolant loop.

SECTION 4

AERODYNAMIC CONTROL SYSTEM

Each vehicle element contains an aerodynamic flight control system used to control the vehicle subsequent to atmospheric entry. Primary flight controls include elevons, ruddervators, and wing spoilers. Secondary flight control is supplied by wing trailing-edge flaps.

Three independent hydraulic systems supply power. Three hydraulic actuators controlled by a triplex hydraulic valve are used to position each control surface. A description of the hydraulic actuation system is presented in Section 6. Command signals include triple redundancy with associated monitoring for detecting failure conditions. A simplified schematic of the aerodynamic flight control system is presented in Figure 4-1.

DESIGN PARAMETERS

Control surface hinge moments and deflection rates are required to size each hydraulic system. In addition, the duty cycle is necessary for establishing the quantity of fuel required by the auxiliary power unit. The significant design parameters used in establishing the system requirements are presented in Table 4-1.

Table 4-1. Design Parameters

		FR-3	
	FR-4	Booster	Orbiter
Elevon Design Parameters			
Total area, S (ft^2/vehicle)	360	805	222
Mean chord, c (ft)	13.25	13.25	13.25
Angle of Attack, α (deg)			
Entry	15	15	15
Cruise	3	3	3
Approach and Landing	5	5	5
Surface Deflection from Neutral, δ (deg)			
Entry	±15	±15	±15
Cruise	±20	±20	±20
Approach and Landing	±40	±40	±40

Table 4-1. Design Parameters, Contd

	FR-4	FR-3 Booster	FR-3 Orbiter
Maximum Deflection Rate, $\dot{\delta}$ (deg/sec)	30	30	30
Entry	30	30	30
Cruise	30	30	30
Approach and Landing	30	30	30
Minimum Deflection Rate, $\dot{\delta}$ (deg/sec)			
Entry	4.5	4.5	4.5
Cruise	1.5	1.5	1.5
Approach and Landing	4.5	4.5	4.5
Dynamic Pressure, q (lb/ft^2)			
Entry	300	300	300
Cruise	184	184	184
Approach and Landing	150	150	150
Duty Cycle, Time at Maximum Rate (%)			
Entry	10	10	10
Cruise	1	1	1
Approach and Landing	10	10	10
Elevon Hinge Moment, T_{elev} (lb-ft/vehicle)			
Entry	358,000	800,000	220,000
Cruise	134,000	300,000	83,000
Approach and Landing	358,000	800,000	220,000
Ruddervator Design Parameters			
Total Area, S (ft^2/vehicle)	600	658	351
Mean Chord, c (ft)(35% × 26.5)	9.26	9.26	9.26
Hinge Sweep Angle, θ_{HSW} (deg)	39.1	39.1	39.1
Dihedral Angle, θ_{DHD} (deg)	45	45	45
Angle of Attack, α (deg)			
Entry	15	15	15
Cruise	3	3	3
Approach and Landing	5	5	5
Surface Deflection from Neutral, δ (deg)			
Entry	±15	±15	±15
Cruise	±20	±20	±20
Approach and Landing	±40	±40	±40

Table 4-1. Design Parameters, Contd

	FR-4	FR-3 Booster	FR-3 Orbiter
Maximum Deflection Rate $\dot{\delta}$ (deg/sec)			
Entry	30	30	30
Cruise	30	30	30
Approach and Landing	30	30	30
Minimum Deflection Rate, $\dot{\delta}$ (deg/sec)			
Entry	4.5	4.5	4.5
Cruise	1.5	1.5	1.5
Approach and Landing	4.5	4.5	4.5
Dynamic Pressure, q (lb/ft^2)			
Entry	300	300	300
Cruise	184	184	184
Approach and Landing	150	150	150
Duty Cycle, Time at Maximum Rate (%)			
Entry	10	10	10
Cruise	1	1	1
Approach and Landing	10	10	10
Ruddervator Hinge Moment, T_{Rud} (lb-ft/vehicle)			
Entry	204,000	224,000	119,000
Cruise	42,000	46,000	24,600
Approach and Landing	244,000	267,000	142,000
Spoiler Design Parameters			
Total Area, S (ft^2/vehicle)	42.1	68	36.5
Mean Chord, c (ft) (10% chord)	1.2	1.2	1.2
Wing Angle of Incidence, i (deg)	6	6	6
Angle of Attack, α (deg)	5	5	5
Deflection Angle, δ (deg)	20	20	20
Maximum Deflection Rate, $\dot{\delta}$ (deg/sec)	30	30	30
Dynamic Pressure, q (lb/ft^2)	150	150	150
Spoiler Hinge Moment, T_{Spoil} (lb-ft/vehicle)	630	1,050	050

Table 4-1. Design Parameters, Contd

	FR-4	FR-3 Booster	FR-3 Orbiter
Wing Flap Design Parameters			
Total Area, S (ft^2/vehicle	410	660	355
Mean Chord, c (ft) (25% chord)	3.3	3.3	3.3
Wing Angle of Incidence, i (deg)	6	6	6
Angle of Attack, α (deg)	4	4	4
Deflection angle, δ (deg)	25	25	25
Maximum Deflection Rate, $\dot{\delta}$ (deg/sec)	2.5	2.5	2.5
Dynamic Pressure, q (lb/ft^2)	150	150	150
Wing Flap Hinge Moment, T_{Flap} (lb-ft/vehicle)	67,000	107,000	58,000

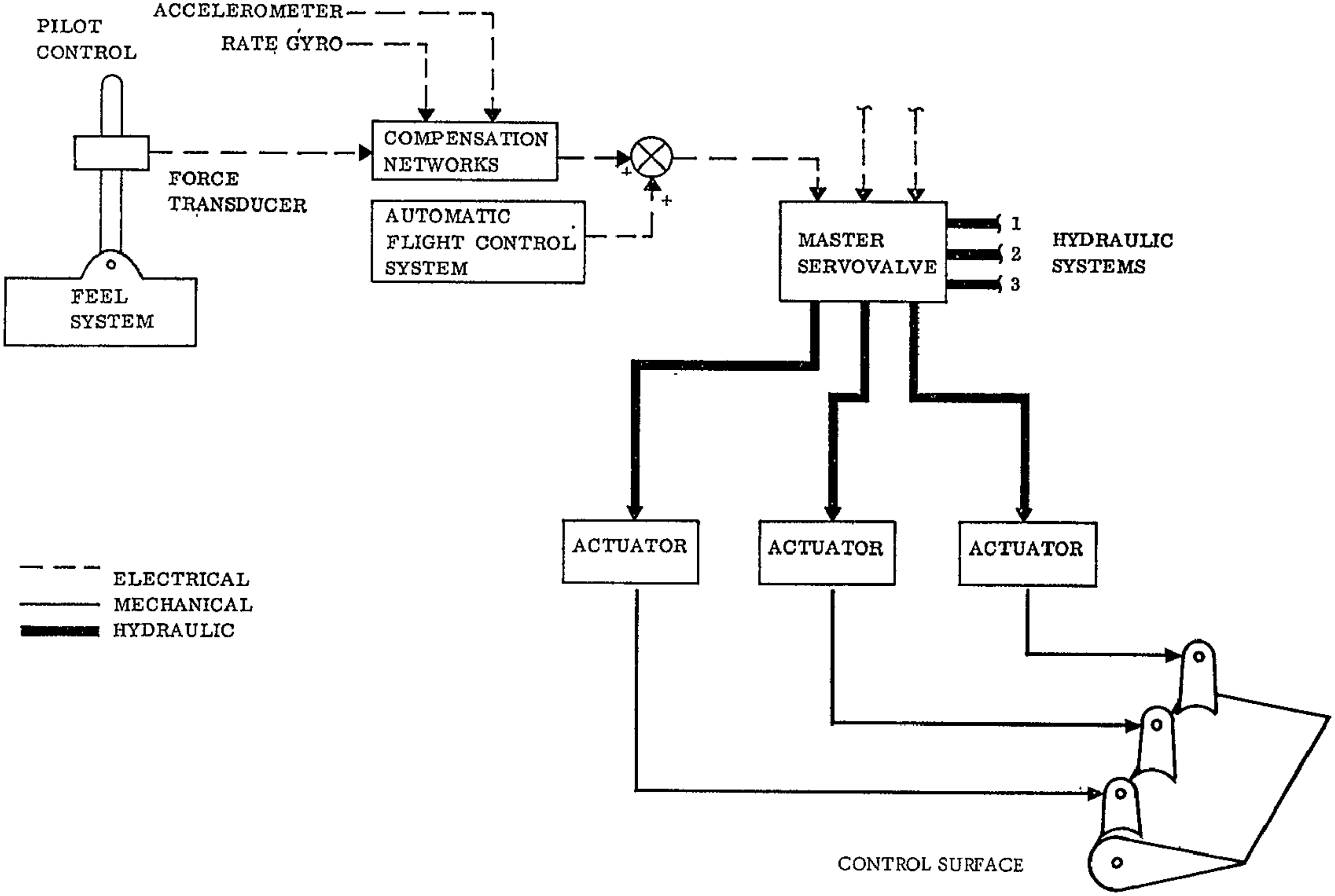

PILOT
CONTROL
ACCELEROMETER
RATE GYRO
COMPENSATION NETWORKS
FORCE TRANSDUCER
AUTOMATIC FLIGHT CONTROL SYSTEM
+
+
MASTER SERVOVALVE
1
2
3
HYDRAULIC SYSTEMS
FEEL SYSTEM
ACTUATOR
ACTUATOR
ACTUATOR
ELECTRICAL
MECHANICAL
HYDRAULIC
CONTROL SURFACE

Figure 4-1. Typical Aerodynamic Flight Control System Schematic

SECTION 5

ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM

5.1 GROSS SYSTEM REQUIREMENTS

This section sets forth the space shuttle requirements and constraints that provide direction for concepts and designs of the Environmental Control/Life Support System (EC/LSS).

5.1.1 CREW COMPARTMENT, ORBITER. The basic requirements is to provide shirtsleeve comfort and safety for a crew of two men in missions having atmospheric and space flight durations as follows:

Liftoff to 50 n.mi. Altitude	6 minutes
Space Activities, Zero-g	1 to 7 days
Atmosphere Entry and Descent, Supersonic	41 minutes
Subsonic Cruise, Approach and Landing	15 minutes

The orbiter has surfaces in which space radiators may be incorporated, but their use for heat rejection shall not impose orientation requirements on the vehicle.

5.1.2 CREW COMPARTMENT, BOOSTER. The basic requirement is to provide shirtsleeve comfort and safety for a crew of two men in missions of the following durations:

Boost and Staging	2.8 minutes
Entry, Deploy Wings and Turbojet Engines	6.1 minutes
Subsonic Cruise in Atmosphere	50 minutes
Descent	15 minutes
Approach and Landing	5 minutes
Total	78.9 minutes

5.1.3 GENERAL REQUIREMENTS FOR THE EC/LSS. The EC/LSS will provide the functions indicated in Table 5-1 for the booster and orbiter modules.

The orbiter cabin atmosphere must be compatible with that of the space station/base, which is 10 psia. The O_2 partial pressure is 2.7 psi and the diluent is N_2. Other significant parameters are:

	Booster	Orbiter
Atmosphere Leakage, lb/day		3.5
Cabin Repressurization per Mission	0	1
Cabin Volume, ft^3	344	344
CO_2 Nominal Limit, mm Hg Partial Pressure	5	5
CO_2 Production, lb/man-day	2.06	2.06
O_2 Consumption, lb/man-day	1.68	1.68

The EC/LSS is required to provide cooling for electrical and electronic equipment for communications and navigation. Items with cold plates are assumed to be 80% effective in rejecting heat to the liquid coolant, with the remaining 20% transmitted to the cabin atmosphere. Liquid coolant is also required for the fuel cell power source of the orbiter. Items within the cabin which do not have cold plates reject heat to the cabin atmosphere, from which it is transferred to liquid coolant.

As with other equipment, the EC/LSS must incorporate capability for preflight and inflight onboard checkout and for expeditious postflight refurbishment. Minimum weight and volume are important objectives in selecting EC/LSS concepts and designs. Minimizing electrical power demand is also important due to effects on the power source and its expendable reactants, and effects on the heat rejection subsystem which may have expendable coolants. Liquid wastes from the crew may be dumped overboard, but must be stored long enough to provide choice of time and place of dumping.

Table 5-1. EC/LSS Functions

	Booster	Orbiter
Atmosphere Supply and Pressurization Control		
Atmosphere Stores	x	x
Atmosphere Pressurization Control	x	x
Atmosphere Purification		
Humidity Control	x	x
Carbon Dioxide Control	x	x
Trace Contaminant Control		x
Pressure Suit Loop		x
Water Management	x	x
Food Management		x
Personal Hygiene		x
Waste Management		x
Thermal Control		
Cabin Atmosphere Cooling	x	x
Cabin Atmosphere Heating		x
Equipment Cooling	x	x
Heat Transport	x	x
Heat Rejection	x	x

5.2 TRADE STUDIES

5.2.1 CREW COMPARTMENT, ORBITER. Point design trade studies were made for the heat sink, CO_2 control, atmospheric stores, sensible heat control, and waste management. They are based on a flight duration of seven days.

5.2.1.1 Heat Sink. Methods were compared on the following basis:

a. The cabin equipment rejects heat to a heat transport fluid, which then rejects it to a heat sink. The heat transport fluid is water with a corrosion inhibitor. Water temperatures to and from the heat sink are 100°F and 35°F, respectively.

b. The design point heat load is 10,000 Btu/hr.

c. Candidate methods for space flight heat rejection are a radiator, a water boiler, and a water sublimator.

d. Candidate methods for atmospheric heat rejection are a ram air heat exchanger, heat transfer to the turbojet engine fuel, expendable cryogenic nitrogen, and expendable Freon.

e. There is 100% redundancy in liquid heat transport circuits. This requirement may be met in radiators by redundant tubing which shares common fins.

f. The fuel cell power source will produce 652 lb of water, of which 452 lb is excess to needs for crew consumption and may be used expendably for heat rejection.

Water Sublimator[5-1, 5-2]. The sublimator has water on one side of a porous plate and space vacuum on the other. In operation, the pores fill with water which freezes to ice on the outlet side. The ice sublimes to space. A pore vacated by sublimation thereupon refills with water and the cycle repeats. Water feed is self-regulating. Assuming that water is supplied at 70°F (h_f = 38 Btu/lb), and freezing is followed by sublimation to saturated vapor at 32°F (h_g = 75.8 Btu/lb), the heat sink capacity is 1037.8 Btu/lb. Estimated weights for the subsystem are:

Water	
Heat rejection @ 10,000 Btu/hr	1620 lb
Contingency, 5%	81
Subtotal	1701
Fuel Cell Water Available	452
Stored Water Requirement	1249
Hardware	
2 Sublimators (One Redundant)	52 lb
Water Tankage @ 16% of Stored Water	200
Subtotal	252

Plumbing and Supports @ 15% of Hardware	38
Hardware Total	290
Line Fill and Residuals @ 1 ft^3	62
Total-Non-expendable	352
Total Expendable	1249
Total Weight	1601 lb

Very little electrical power is required for controls and measurements, and for the increment in pumping power due to sublimator pressure drop. The estimated volume is 27 ft^3.

Water Boiler[5-1]. Operation of a water boiler is similar to that of a sublimator except that back pressure is controlled to prevent freezing. Control of water feed is also required. Performance of water boilers is reported to be less satisfactory than sublimators, particularly with fluctuating heat loads. One result is carry-over of liquid to space without performing useful cooling. Weight estimates are:

Water	
Heat Rejection @ 10,000 Btu/hr	1620 lb
Carry-over Allowance @ 10%	162
Subtotal	1782
Contingency, 5%	89
Subtotal	1871
Fuel Cell Water Available	452
Stored Water Requirement	1419
Hardware	
2 Water Boilers (one redundant)	39 lb
Water Tankage @ 16% of Stored Water	227
Subtotal	266
Plumbing and Supports @ 15% of Hardware	40
Hardware Total	306
Line Fill and Residuals @ 1 ft^3	62
Total Non-expendable	368
Total Expendable	1419
Total Weight	1787 lb

Very little electrical power is required for controls and measurements, and for the increment in pumping power due to pressure drop in the boiler. The estimated volume is 31 ft^3.

Space Radiators. The performance of radiators is so highly sensitive to thermal and geometric parameters that only a generalized evaluation can be made. Since the radiator is not to constrain the vehicle attitude, the performance must be based on the worst case among all possible vehicle attitudes and orbit positions.

A candidate concept is that the radiator is fixed to the vehicle on outer surfaces. By wrapping the radiator at least part way around the fuselage, or by using both sides of the fins, the combination of area and multi-directional capability would achieve sufficient heat transfer that attitude orientation would not be required. These locations have the serious disadvantages that (1) radiator thermal coatings would be damaged by entry heating, (2) structural penalties are involved, and (3) rocket exhaust impingement from attitude control engines would impair radiator performance and damage thermal control coatings.

Another candidate concept is of radiators which are stowed within the fuselage for protection during boost and entry, and are deployed during space flight. Recent studies[5-3] were made of deployed radiators in configurations involving multiple docked vehicles in earth orbit at 260 n.mi. Thermal performance was determined considering the radiant interchange of the various surfaces, which included the several vehicles, solar cell arrays, and other radiators. Excluding configurations with the radiators attached to solar arrays (therefore receiving unfavorable attitude orientation), the heat rejection rate was in the range of 39 to 49 Btu/hr-ft^2.

Since the orbiter wings may be stowed during boost and entry, their surfaces receive the protection that is required for a radiator, and they could be deployed in space flight for that purpose. By using both upper and lower surface, sufficient multi-directional capability appears attainable so that no orientation is required. Studies of flat panel radiators[5-4] indicate that the minimum net heat transfer will be about 30 Btu/hr-ft^2 of panel with both surfaces exposed (two ft^2 of surface rejecting heat).

For the purposes of this trade study it will be assumed that the most unfavorable attitude and orbit position will limit heat rejection to 25 Btu/hr-ft^2. The required panel area for 10,000 Btu/hr is then 400 ft^2. Installed in a wing, this requires 400 ft^2 of planform area with 400 ft^2 of radiator in each the upper and lower surfaces. Planform area available (a minimum of 800 ft^2) greatly exceeds this requirement.

Estimated weights for radiators are in the range of 0.5 to 1.25 lb/ft^2. The extent is not yet determined to which weight economy can be attained by exploiting structural properties of a radiator in wing design. This trade study uses 0.5 lb/ft^2 for 800 ft^2 of surface area, which is equivalent to 1 lb/ft^2 for 400 ft^2 of wing planform area. The radiator system estimated weights are based on redundant plumbing in the same manner as the water boilers and sublimators. The radiator fluid will be one of very low freezing point to prevent freeze-up in the radiator under low-load conditions. Candidate control methods, subject to detailed thermal analysis, include intermittent bypass, modulating bypass, regenerative heat exchange, and combination regenerative-bypass. Estimated weights are:

Radiator Panels	400 lb
Fluid Fill	90
Pumping Unit, Radiator Fluid	25
Radiator Controls	35
Heat Exchangers, HTF to Radiator Fluid	20
Plumbing, EC/LSS to Wing Area	20
Total	590 lb

The estimated radiator fluid pumping power is 80 watts. Volume required in the fuselage is estimated at 3 ft^3.

Ram Air Heat Exchangers. A ram air heat exchanger would be an effective heat sink only during subsonic flight. The orbiter descent profile has a period of 41 minutes in atmospheric entry and only 15 minutes in subsonic flight. Therefore, a ram air heat exchanger would not be useful for sufficient time to justify the application.

Turbojet Fuel. For this trade study the required heat sink capacity for turbojet fuel is based on a heat load of 10,000 Btu/hr for a maximum time of 1.5 hour. The turbojet engines are operated only during the final few minutes of flight, so that the entire stored quantity is available for heat sink during most of the descent time.

For a quantity of 21,000 lb of JP-4, a heat load of 10,000 Btu/hr, and a specific heat of 0.45 Btu/lb-°F, the temperature rise is only 1.59°F. However, the fuel would have to be several degrees colder than the 35°F stated for the EC/LSS heat transport fluid. One approach is to chill the JP-4 to about 32°F in a ground facility and to fuel the vehicle shortly before launch. It is expected that the thermal design could result in adequate insulation and isolation to prevent significant temperature rise of the fuel during seven days of space flight. Heat transfer within the liquid under zero-g would be only by diffusion.

Another approach is to load the JP-4 at normal ground temperature and to use it for heat sink at the end of the mission with no thermal conditioning. The result would be degraded thermal performance of the EC/LSS, which might be acceptable as a transient condition for the short duration of the entry, descent, and landing. Estimated weights for hardware to use either JP-4 heat sink approach, with 100% redundancy are:

JP-4 Pumps with Filters and Check Valves	12 lb
Heat Exchangers, Heat Transport Fluid to JP-4	20
Plumbing	6
Fluid Fill	20
Total	58 lb

Estimated volume of the installed hardware is 2 ft^3 and estimated electrical power for pumping the JP-4 is 30 watts.

Subcritical Nitrogen. The thermal capacity of N_2 from 150 psia subcritical storage to vapor at 14.7 psia and 60° F is 158 Btu/lb. Adiabatic flow through an expansion valve over the pressure range indicated would produce mixed phase fluid of 27% quality and a temperature of -320° F. Adequate control may require staged pressure reduction and interstage heat transfer. Assuming the conditions stated and that N_2 vessel pressurization is maintained thermally by the heat being rejected, the entire 158 Btu/lb capacity is useful. Estimated weights for the heat sink system are:

Cryogenic Nitrogen	
Heat Rejection, 10,000 Btu	63 lb
Contingency, 5%	3
Subtotal	66
100% Redundancy	66
Stored N_2 Requirement	132 lb
Hardware	
2 Heat Exchangers (One Redundant)	20
2 Subcritical N_2 Vessels	52
Plumbing, Supports	11
Total Hardware	83
N_2 Residuals	5
Total Non-expendable	88
Total Expendable	132
Total Weight	220 lb

A very small amount of electrical power is required for controls and measurements. The estimated volume is 5 ft^3.

Liquid Freon-22. The thermal capacity of Freon-22 from liquid at 70° F and 137 psia to vapor at 60° F and 14.7 psia is 85 Btu/lb. Adiabatic expansion would produce mixed phase fluid at -41° F. Estimated weights for a heat sink system are:

Freon-22	
Heat Rejection, 10,000 Btu	128 lb
Contingency, 5%	6
Subtotal	134
Redundancy, 25%	34
Stored F-22 Requirement	168 lb
Hardware	
2 Heat Exchangers (One Redundant)	22 lb
F-22 Tanks @ 10% of Stored Liquid	17
Subtotal	39
Plumbing and Supports @ 15% of Hardware	6
Total Hardware	45
Total Expendable	168
Total	213 lb

A very small amount of electrical power is required for controls and measurements. The estimated volume is 3 ft^3.

Summary of Orbiter Heat Sink Trade Study. Characteristics of the candidate heat sinks for space flight are shown in the following table.

Heat Sink	Weight (lb)	Volume (ft^3)	Power (watts)
Water Sublimator	1601	27	Negligible
Water Boiler	1787	31	Negligible
Radiator	590	3	80

Water boilers are least acceptable because of high weight and volume, and because they have not been completely satisfactory in previous flight experience. The water sublimator could be chosen with confidence based on very satisfactory previous flight experience, but the weight and volume are quite high. The radiator approach is recommended due to very much lower weight and volume. However, confidence in the radiator evaluation is lower than confidence in the water sublimator evaluation. Accurate radiant interchange data are not available without comprehensive analysis of final configurations together with firm orbit and attitude parameters. Accurate weight data will depend on detailed thermal and structural design of the proposed radiators on wing surfaces.

Characteristics of the orbiter candidate heat sinks for atmospheric flight are:

Heat Sink	Weight (lb)	Volume (ft^3)	Power (watts)
Ram Air Heat Exchanger		Not Applicable	
Turbojet Fuel	58	2	30
Cryogenic Nitrogen	220	5	Negligible
Liquid Freon-22	213	3	Negligible

Problems of control and extreme temperature differences are antitipated for a cryogenic N_2 heat sink. The Freon-22 would be significantly better in these characteristics. The turbojet fuel heat sink is recommended due to minimum weight and volume. Probably more important is that it has greater capacity than the other candidates as backup to the radiator in space flight.

5.2.1.2 Carbon Dioxide Control. The candidate methods for carbon dioxide control are cabin purging, absorption in lithium hydroxide, and adsorption in regenerable molecular sieves.

To remove 2.06 lb/man-day of CO_2 from a 10 psia atmosphere maintaining 5 mm Hg CO_2 partial pressure will require a purge of 137.5 lb/man-day. The loss of useful gases for two men and 7 days is:

O_2	573 lb
N_2	1298
H_2O	23
Total	1894 lb

Lithium Hydroxide. Estimated weights, based on Apollo hardware are:

CO_2 Absorber Canister (dual), Apollo P/N 811400	17.7 lb
CO_2 Absorber Filters, Apollo P/N 811510 Capacity 36 man-hours each 10 filters @ 4.7 lb each	47.0
Total	64.7 lb

The volume required is about 3 ft^3.

Molecular Sieves. A characteristic of molecular sieves is that water vapor must be pre-adsorbed from the atmosphere prior to flow into synthetic zeolite for CO_2 adsorption. In one type of regeneration the water is lost because space vacuum is used to desorb both the water adsorber and the CO_2 adsorber. In another type, which is more complex, the water is desorbed into a return flow of cabin atmosphere. Weight estimates for a 2-man unit are:

	Water Loss Type	Water Save Type
Installed Unit, lb	110	165
Redundant Unit, lb	110	165
Water loss, 28 Man-Days, lb	24	
Total	244	330

Estimated volume for either type is 7 ft^3.

Summary of Orbiter CO_2 Control Trade Study. The weight penalty for cabin purge is excessive. Molecular sieves are complex and have relatively high weight and volume. Therefore, lithium hydroxide is recommended for CO_2 control.

5.2.1.3 Atmospheric Stores. The candidate methods are high pressure gas, cryogenic supercritical, and cryogenic subcritical. The trade study if based on the following quantities:

	Oxygen (lb)	Nitrogen (lb)
Crew Consumption	23.5	
Leakage	7.3	16.5
Repressurization	5.2	11.9
Subtotal	36.0	28.4
Redundancy	18.0	
Total	54.0	28.4

Storage vessel weights and volumes for O_2 are based on:

High pressure gas: $W_v = 10 + 1.42\, W_u$ lb

$V_v = 0.0436\, W_u + 0.3\ \text{ft}^3$

Supercritical $W_v = 17 + 0.205\, W_u$ lb

$V_v = 0.0228\, W_u + 0.5\ \text{ft}^3$

Subcritical: $W_v = 17 + 0.151\, W_u$ lb

$V_v = 0.0228\, W_u + 0.5\ \text{ft}^3$

where

W_u = weight of usable fluid

W_v = weight of storage vessel, lb

V_v = volume of storage vessel, ft^3

The expressions for N_2 are:

High pressure gas: $W_v = 10 + 1.62\, W_u$ lb

$V_v = 0.05\, W_u + 0.3\ \text{ft}^3$

Supercritical $W_v = 17 + 0.29\, W_u$ lb

$V_v = 0.0323\, W_u + 0.5\ \text{ft}^3$

Subcritical: $W_v = 17 + 0.214\, W_u$ lb

$V_v = 0.323\, W_u + 0.5\ \text{ft}^3$

Following is a summary of vessel weights and volumes from the above relationships:

	Oxygen		Nitrogen	
	Weight (lb)	Volume (ft^3)	Weight (lb)	Volume (ft^3)
High Pressure Gas	87	2.7	56	1.7
Supercritical	28	1.7	25	1.4
Subcritical	25	1.7	23	1.4

The weight and volume differences between supercritical and subcritical are relatively small. Supercritical is simpler and more reliable in quantity gaging and in control. Therefore, supercritical storage is recommended for both oxygen and nitrogen.

5.2.1.4 Sensible Heat Control. Candidate methods for cooling the cabin atmosphere are: (1) circulation of the atmosphere through a conventional liquid-cooled heat exchanger, and (2) circulation against cold walls. Candidate methods for cooling equipment are: (1) liquid-cooled cold plates, and (2) circulation of cabin atmosphere. Candidate methods for removing crew sensible heat are (1) circulation of cabin atmosphere, and (2) liquid-cooled garments. For this trade study, the estimated sensible heat load to the cabin atmosphere is 4500 Btu/hr and cabin temperature is to be selectable from 65° to 75°F with a control accuracy of ±2°F.

Gas-to-Liquid Heat Exchangers. Based on prior studies with similar requirements[5-5, 5-6, 5-7] the estimated weights are:

2 Heat Exchangers (One Redundant) w/fluid	35 lb
2 Blowers (One Redundant)	10
Controls	3
Installation Hardware	12
Total	60 lb

Estimated blower power is 80 watts and volume is 5 ft^3.

Cold Walls. Transferring all sensible heat through the walls eliminates the sensible heat exchangers and reduces the load on the liquid heat transport and heat rejection components. Since the vehicle is not to be constrained in attitude for thermal control purposes, the exterior surfaces of the walls may be exposed to extremes in radiant environment. This produces severe control problems.

The interior conditions are constrained by the requirements for closely controllable (±2°F) atmosphere temperature, and for surface temperatures above the dew point so that uncontrolled condensate will not form. Thermal design of the wall, as to insulating techniques and selection of exterior surface coatings, must provide high enough heat efflux to equal the maximum sensible heat load at the highest temperature environmental heat sink. The same design must limit heat efflux to the minimum sensible heat load at the lowest temperature environmental heat sink. These conditions are incompatible without an active control.

A small degree of control is attainable by varying the rate of atmosphere circulation. Limitations on high rate come from noise, high electrical power, and a chance of uncomfortably high velocities impinging on the crew. Lowering the circulation rate is associated with control to reduce heat efflux by increasing the film thermal resistance. This would increase the atmosphere-to-wall temperature difference. Since the atmosphere temperature is fixed by comfort requirements, the wall temperature must decrease, and it is dew point limited. Therefore, the control range is limited.

Thermal control louvers may have sufficient range for the environmental extremes anticipated. An estimate for cooling electronic equipment[5-8] is 1 ft^2 of interior louver weighing 1.5 lb per 40 watts of thermal radiation. Considering probable temperatures of the cabin atmosphere, the interior and exterior surfaces of the cabin walls, and the environmental heat sink, the heat transfer rate will be much lower. An estimate of 20 Btu/hr-ft^2 is based on an analysis of panels at various attitudes in low earth orbit[5-4]. For a sensible heat load of 4500 Btu/hr, the required wall surface area is 225 ft^2. Assuming that control by louvers is applied to the total area, the weight is 337 lb for control only.

A full evaluation of the cold wall concept would require extensive analysis. Performance is highly sensitive to thermal and geometric parameters involving vehicle attitude and orbit position, and the thermal design of the crew compartment.

Equipment Sensible Heat Control. The selection of liquid-cooled cold plates versus atmospheric cooling is usually dependent on the type of component. Lights, displays, fans, controls and measurement components are normally of characteristics such that cold plating would be cumbersome and atmospheric cooling is entirely satisfactory. Certain other components may benefit significantly from the liquid cooling characteristics of higher thermal inertia and higher heat transfer rates per unit area and volume.

If a component must maintain a critical function in the absence of an atmosphere, liquid or radiation cooling is mandatory.

When the selection is not obvious for a particular component, a weight trade study can be made. The thermal control subsystem will incur a weight penalty of about 0.03 lb/Btuh for heat transferred to the cabin atmosphere versus transfer to the liquid coolant. This must be traded against the weight increment due to applying a cold plate to the component.

Crew Sensible Heat Control. Liquid-cooled garments are not compatible with the shirtsleeve attire requirement. Therefore, the crew will be cooled by circulated cabin atmosphere.

Summary of Orbiter Sensible Heat Control Trade Study. It is not necessary that sensible heat control be accomplished exclusively by one method. The following is an approach to an optimum design under the anticipated wide range in the thermal radiation environment of the vehicle.

a. Design thermal control coatings for the exterior walls such that there is little or no net heat influx in the worst heating environment. It may be possible to always provide an efflux. This will minimize penalties to the heat transport and heat rejection components.

b. Design wall thermal resistance to limit heat efflux in the worst cooling environment. This limit is determined by maintaining all inside wall surfaces above the dew point.

c. For shirtsleeve crew comfort, circulate cabin atmosphere through liquid-cooled heat exchangers, using an anticipating type control with selectable control point. Maintain ventilation velocities over the crew in the 15 to 40 ft/min range.

d. If analysis shows conditions that require heating the cabin atmosphere, use the same heat exchanger with control on the liquid side to select hot instead of cold heat transport fluid. A heating requirement is unlikely.

e. Analyze electrical and electronic components on an individual basis to optimize atmospheric versus liquid cooling.

5.2.1.5 Waste Management. The candidates are classified as active and passive systems.

a. Active Waste Management Subsystems. In a trade study for missions from 12 to 60 man-days[5-4] the recommended waste management system employs flow of the cabin atmosphere to assist the zero-g collection of urine and feces. The atmosphere returns to the cabin through an activated charcoal filter for odor control.

   Urine and other liquid wastes are jettisoned through an Apollo-type disposal lock. Feces and other solid wastes are collected, dried, and stored in a single container, which is sealed from the cabin and exposed to space vacuum when not in use for collection. It is estimated that the system will have a dry weight of 56 lb, a volume of 3 ft^3, and a peak power demand during collection of 80 watts.

b. Passive Waste Management. Plastic bags can be used for urine collection and defecation gloves for feces collection. Both involve some hazard to sanitation, no control of odor, and unpleasant manual handling of wastes. Following collection, the wastes can be manually mixed with a disinfectant to inhibit decomposition and production of noxious gases in storage. Due to sanitation hazards and crew time required, passive waste management is not recommended except as a backup to an active system.

5.2.2 CREW COMPARTMENT, BOOSTER. Trade studies for the booster crew compartment are considered separately from the orbiter because of the different flight profile and mission duration (1.3 hours).

5.2.2.1 Heat Sink. Methods were compared on the following basis:

a. The cabin equipment rejects heat to a heat transport fluid, which then rejects it to the heat sink. The heat transport fluid is water with a corrosion inhibitor. Water temperatures to and from the heat sink are 100° and 45°F, respectively.

b. The design point heat load is 6500 Btu/hr.

c. Candidate methods are a ram air heat exchanger, heat transfer to the turbojet engine fuel, expendable cryogenic nitrogen, and expendable Freon.

d. There is 100% redundancy in heat transport circuits.

Ram Air Heat Exchanger. The booster has flight durations of about eight minutes in boost and entry and 70 minutes in subsonic flight. Therefore, a ram air heat exchanger could be an effective heat sink during most of the flight. It is estimated that thermal inertia would suffice for the eight minutes prior to subsonic flight. Considering thermal mass of the heat transport fluid only, the temperature rise is 14°F. Estimated characteristics of the system are:

2 Ram Air Heat Exchangers (One Redundant)	2 lb
Fluid Fill	4
Mounting	1
Plumbing, ECS to Ram Air Heat Exchangers	4
Total	11 lb

Very little electrical power is required for controls, measurements, and the increment in pumping power due to pressure drop in the heat exchanger. Estimated installed volume is 1 ft^3.

Since the ram air heat exchangers should be deployed at the same time as the engines, they can be mounted on the same deployment mechanisms if geometry permits. A calculation of core face area shows less than 0.15 ft^2, which is small enough that a potential location is within the engine pylons.

Turbojet Fuel. The comments for the orbiter contained in Paragraph 5.2.1.1 are applicable to the booster. Chilling the JP-4 to about 32°F prior to flight is a particularly effective method because use of the fuel for heat sink can start before liftoff and continue uninterrupted until the end of flight.

Considering that the hardware should be the same as for the orbiter, the weight, volume, and power estimates are the same; i.e., 58 lb, 2 ft^3, and 30 watts.

Subcritical Nitrogen. The comments for the orbiter contained in Paragraph 5.2.1.1 are applicable but the total heat to be rejected is less. With the same hardware, but less expendable cryogen required, the weights are:

Total non-expendable	88 lb
Total expendable	112
Total weight	200 lb

Liquid Freon-22. Adjusting expendables from Paragraph 5.2.1.1 to the booster heat load, the weights are:

Total hardware	45 lb
Total expendable	143
Total weight	188 lb

Summary of Booster Heat Sink Trade Study. Characteristics of the candidate heat sinks are:

Heat Sink	Weight (lb)	Volume (ft^3)	Power (watts)
Ram Air Heat Exchanger	11	1	Negligible
Turbojet Fuel	58	2	30
Cryogenic Nitrogen	200	5	Negligible
Liquid Freon-22	188	3	Negligible

The expendable heat sinks are high in weight and volume. Using turbojet fuel imposes a requirement to chill it before fueling the booster. Ram air heat exchangers are recommended due to minimum weight and minimum requirements for ground operations.

5.2.2.2 CO_2 Control. During 1.3 hours of flight the CO_2 added by two men in 344 ft^2 will rise the partial pressure to 4.4 mm Hg. This is within the specified limit in Paragraph 5.1.3. Therefore, no CO_2 control is required.

5.2.2.3 Atmospheric Stores. It is assumed that air will be bled from the cabin during boost, ending at a regulated pressure of 10 psia. To restore sea level O_2 partial pressure will require an inflow of about two pounds. Gaseous storage is preferred for this small quantity. Leakage at 3.5 lb/day for 1.3 hours from a cabin initially at 14.7 psia will reduce the pressure to 14.6 psia. Therefore, a no atmospheric stores are required to make up leakage.

Oxygen consumed by two men for 1.3 hours at 1.68 lb/man-day will reduce O_2 partial pressure from 3.1 psi to 3.0 psi. Therefore, no atmospheric stores are required to make up O_2 consumed. It is recommended that the crew compartment be equipped with standard aircraft emergency oxygen bottles and face masks for use if a severe leak occurs during the few minutes of flight outside the atmosphere.

5.2.2.4 Sensible Heat Control. The discussion on sensible heat control for the orbiter is applicable to the booster, except that radiant heat rejection from exterior wall surfaces will not be significant.

5.2.2.5 Waste Management. The flight duration is short enough that waste management is not required for the booster.

5.3 DETAIL SYSTEM REQUIREMENTS

This section presents requirements supplemental to and in more detail than discussed in Section 5.1 and 5.2.

5.3.1 CREW COMPARTMENT, ORBITER. Shirtsleeve conditions require control within the ranges shown:

Cabin Atmosphere, psia	10 ±0.2
Temperature, Selectable, °F	65 to 75 ±2
Relative Humidity, %	30 to 60
Ventilation Velocity Over Crew, fpm	15 to 40
Oxygen Partial Pressure, psi	2.7 +0.4, -0.0

5.3.1.1 Cabin Atmosphere. Quantity requirements for the cabin atmosphere are based on properties at average conditions of 70° F and 50% relative humidity, as listed in Table 5-2.

Table 5-2. Cabin Atmosphere Requirements

Gas	Partial Pressures (mm Hg)	Partial Pressures (psi)	Density (lb/cu ft)	Fractions By Weight	Fractions By Volume
O_2	139.5	2.700	0.0152	0.298	0.2700
N_2	363.2	7.021	0.0346	0.675	0.7021
N_2O	9.3	0.182	0.00059	0.012	0.0182
CO_2	5.0	0.097	0.00075	0.015	0.0097
Total	517.0	10.000	0.05114	1.000	1.0000

The stored O_2/N_2 requirement is given in Section 5.2.1.3; the requirement for the CO_2 is given in Section 5.2.1.2.

5.3.1.2 Water Management. Based on Reference 5-1. a water balance for the crew, in lb/man-day, is given in Table 5-3.

Table 5-3. Crew Water Balance

	Input	Output
Contained in Food and Drink	6.99	
Metabolically Formed	0.78	
Contained in Urine		3.08
Contained in Feces		0.25
Contained in Food Wastes		0.14
Respiration and Perspiration		4.30
Total	7.77	7.77 lb/man-day

The mission water balance is given in Table 5-4.

Table 5-4. Mission Water Balance

	Supply (lb)	Consumption (lb)
Humidity Condensate	60	
Fuel Cell Water Production	652	
Personal Hygiene		32
Urinal Flush		28
Drinking and Food Preparation		98
Subtotal	712	158
Excess		554
Total	712	712

Urine and used wash water are dumped overboard after temporary storage to permit choice of time and place. Assuming storage not more than two days, the required capacity is:

Urine and Urinal Flush Water	20.3 lb
Used Wash Water	9.2 lb
Total	29.5 lb

A thermal control water sublimator may be advantageous to supplement the space radiator for brief peak loads or brief periods of unfavorable thermal radiation environment. The excess fuel cell water may be stored for use as an expendable heat sink in the sublimator. Storage of 2 ft^3 (about 120 lb) would provide a heat sink capability of about 120,000 Btu.

5.3.1.3 Thermal Control. Heat loads originating in the EC/LSS are based on estimated electrical inputs as presented in Table 5-5.

Table 5-5. EC/LSS Equipment Heat Loads

Subsystem	Component	Power (watts) Average	Power (watts) Maximum	Duty Cycle
Thermal Control	Pump, HTF	34	34	Continuous
	Pump, radiator fluid	80	80	
	Cabin, radiator fluid	80	80	Continuous
Atmosphere Purification	Cabin loop blower	70	70	Continuous
	Suit boost blower	0	50	Only if failure in pressure
	Water separators	4	4	Continuous
Atmosphere Supply	Heaters	8	40	20% of time
	Gas sensor and control	10	10	Continuous
Waste Management	Feces collector	1	90	1% of time
	Urine collector	1	90	1% of time
Personal Hygiene	LGS/Blower/Pump	2	90	2% of time
Total		290		

The metabolic heat rates to the cabin atmosphere are:

	Btu/Man-Day	Btu/Hr for 2 Man Crew
Sensible	5,820	485
Latent	4,500	375
Total	10,320	860

The use of LiOH for CO_2 absorption creates an additional latent heat load to the cabin atmosphere of 27 Btu/man-hour latent and 77 Btu/man-hour sensible.

All equipment and metabolic heat loads are reflected in requirements to cool the cabin atmosphere, except for cold-plated equipment which transfers 80% of its rejected heat directly to the liquid heat transport fluid. Table 5-6 is a summary of design point heat loads.

During the brief period that the turbojet engines are in operation there will be additional electrical and electronic equipment items in operation. It is considered that all such items will be located so as to reject their heat to the ambient atmosphere.

Table 5-6. Orbiter Heat Load Summary

Item	To Liquid (watts)	To Cabin Atmosphere (watts)	Total (watts)
Inertial Reference and Autopilot	0	280	280
Dual Computer	320	80	400
Star Tracker	0	10	10
Rendezvous Radar	0	100	100
TACAN	72	18	90
Radar Altimeter	0	28	28
Communications	165	40	205
Lighting and Displays	0	200	200
Multiplex Data	80	20	100
Fuel Cells	1,295	0	1,295
EC/LSS Electrical	94	169	290
Subtotal, watts	2,026	972	2,988
Subtotal Btu/hr	6,920	3,320	10,240
LiOH Latent 54 Btu/hr			
Metabolic Latent 375			
Subtotal Latent	0	429	429
LiOH Sensible		154	154
Metabolic Sensible	0	485	485
Totals, Btu/hr	6,920	4,388	11,308

5.3.1.4 Waste Management. Liquid waste quantities are given in Tables 5-3 and 5-4. Solid wastes will include feces, food residues, food packaging materials, and used paper towels. The estimated total quantity is 7 lb with a volume of 0.5 ft^3. There is a requirement for processing and storage such that odors are contained and the usual concepts of sanitation are not violated.

5.3.1.5 Food Management. Food management estimates are based on Apollo-type foods supplemented by some conventional foods which are selected for compatibility with the zero-g environment. Including packaging, the estimated weight is 2 lb/man-day with a storage volume of 0.1 ft^3/man-day. The totals are 28 lb and 1.4 ft^3.

5.3.1.6 Personal Hygiene. Certain minimum hardware will permit whole-body bathing with a sponge, water, and non-foaming detergent. The water balance in Table 5-4 allows 2.3 lb/man-day. For comfort, the water should be heated to about 100° F

5.3.2 CREW COMPARTMENT, BOOSTER. Oxygen supply and CO_2 removal are discussed in Paragraphs 5.2.2.2 and 5.2.2.3. The booster will not require subsystems for water management, food management, waste management, and personal hygiene. The remaining requirements are for comfort temperature and relative humidity as specified in Paragraph 5.3.1.

The estimated heat loads are lower than for the orbiter (Table 5-6) by the amounts given in Table 5-7.

Table 5-7. Booster Heat Load Reductions

	Watts	Btu/hr
ECS Functions Deleted and Reduced	87	
Computer, Dual to Single	160	
Star Tracker Deleted	10	
Rendezvous Radar Deleted	100	
Fuel Cells Deleted	1295	
Subtotal	1652	5650
LiOH Deleted		208
Total Reduction, Btu/hr		5858

Subtracting this amount from the total in Table 5-6 leaves the booster total heat load at 5450 Btu/hr, of which 3400 Btu/hr is to the cabin atmosphere and 2050 Btu/hr is directly to the heat transport fluid.

5.4 DETAIL DESIGN DATA, ORBITER EC/LSS

This section contains schematics of the EC/LSS subsystems, descriptions of operation, and estimates of weight and volume. Symbols and abbreviations used on the schematics are shown in Tables 5-8, 5-9, and 5-10.

5.4.1 ATMOSPHERE SUPPLY AND PRESSURIZATION CONTROL. Figure 5-1 shows the major components of the subsystem. The O_2 is stored in three vessels, of which one is redundant. Table 5-11 lists other redundancies. The N_2 is stored in two vessels with no redundancy. Figure 5-1 also shows the concept of sensor locations for automatic checkout.

Multiple insulated vessels located external to the cabin store cryogenic O_2 and N_2. Pressure within the vessels is maintained supercritical by a combination of heat leak through the insulation and internal electrical heaters. Uninsulated accumulators for

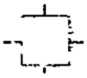
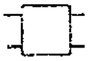
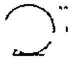
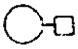
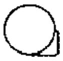
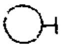
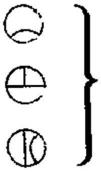
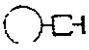
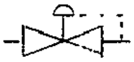
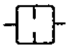
Table 5-8. Abbreviations and Chemical Symbols

A	Accumulator	g	Gage, as in psig
a	Absolute, as in psia	HX	Heat Exchanger
ACF	Activated Charcoal Filter	L	Liquid, as in LO_2
Ag	Silver	LiOH	Lithium Hydroxide
BAC	Benzalkonium Chloride	MF	Multifiltration Unit
D-50	Dowex 50 ion Exchange Resin	WMS	Water Management Subsystem
F	Filter	Po	Position
G	Gaseous, as in GO_2	S	Sterilizer

Table 5-9. Sensor Symbols

Ⓟ	Pressure	Ⓘ	Electrical Current
Ⓣ	Temperature	Ⓥ	Voltage
Ⓠ	Quantity	Ⓒ	Electrical Conductivity
Ⓕ	Flow	(P_o)	Position Indicator
Ⓡ	Rotational Speed		

Table 5-10. Component Symbols

 or 	Heat Exchanger	⊗	Shutoff Valve
	Pump		Remotely-Actuated Valve
	Blower		Manually-Actuated Valve
	Check Valve		Selector Valves
	Relief Valve		Remotely-Actuated Valve with Manual Override
	Pressure Regulator		Flow Restrictor

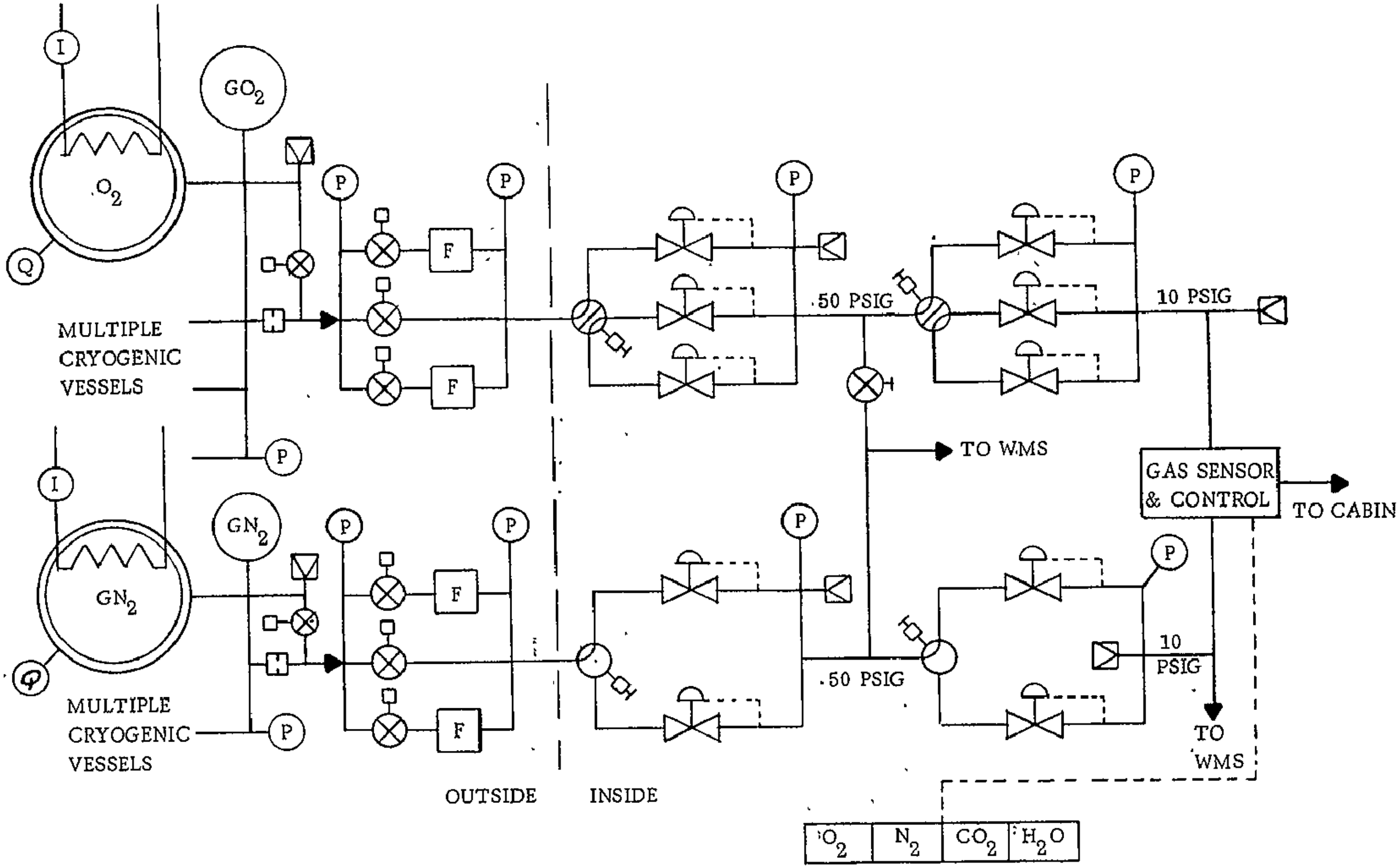

I
GO$_2$
O$_2$
Q
P
F
MULTIPLE
CRYOGENIC
VESSELS
.50 PSIG
10 PSIG
TO WMS
GAS SENSOR
& CONTROL
TO CABIN
I
GN$_2$
GN$_2$
10
PSIG
.50 PSIG
TO
WMS
OUTSIDE
INSIDE
O$_2$
N$_2$
CO$_2$
H$_2$O

Figure 5-1. Atmosphere Supply and Pressurization Control

Table 5-11. Atmosphere Supply and Pressurization Control Redundancies

Component	Redundancies
O_2 Cryogenic Vessels	Multiple vessels, one redundant Redundant heater in each vessel
N_2 Cryogenic Vessels	Multiple vessels, none redundant N_2 content of cabin atmosphere provides back-up
Cabin Pressure Relief Valves	Redundant valves for inflow and outflow relief Manual overrides for inflow and outflow
Gas Filters	Redundant filters Filter bypass
O_2 Pressure Regulators	Two redundant regulators in each stage
N_2 Pressure Regulators	One redundant regulator in each stage N_2 content of cabin atmosphere provides back-up
Gas Sensor and Control	Redundant sensor and control elements Manual overrides
Other Functions	Backup
Cabin Pressure Integrity	Suit loop
Pressurization for WMS	O_2 at 50 psig Separate gaseous N_2 container

the gases provide some capability for rapid flow response. The two-gas pressure control causes withdrawal of the gases from storage such that partial pressures of O_2 and N_2 are maintained within their respective control bands. The flows are from the gas accumulators through components in sequence as follows: (1) a flow restrictor, which prevents flows greater than capacity of the cabin pressure relief valve, (2) a filter, (3) a first-stage pressure regulator, which controls to a downstream pressure of 50 psig, (4) a second-stage pressure regulator, which controls to 10 psig, and (5) the two-gas pressure control, which maintains O_2 partial pressure at 2.7 psi and cabin total pressure at 10 psia.

The cabin pressure relief valve vents gas to relieve cabin pressure greater than 10.5 +0.2 psig. It also provides negative pressure relief when the vehicle is descending into the lower atmosphere.

N_2 is supplied to the water management subsystem at 50 psig and 10 psig for pressurization of expulsion devices.

The weights and installed volumes in Table 5-12 are based on the trade study and on component data in Reference 5-2. All redundancies are included.

Table 5-12. Atmosphere Supply and Pressurization Control Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
Oxygen	83	
O_2 Storage Vessels	34	2.4
Nitrogen	34	
N_2 Storage Vessels	27	1.5
O_2 and N_2 Gas Accumulators	8	1.0
Two-gas Sensor and Control (Dual)	25	0.5
Valves. Filters, Transducers, Plumbing	27	6.4*
Total	238	11.8

*Includes access space for installation and maintenance

5.4.2 ATMOSPHERE PURIFICATION. Figure 5-2 shows the major components of the subsystem, which includes a pressure suit loop as backup to cabin pressurization. Redundancies are summarized in Table 5-13.

Cabin atmosphere is circulated through the atmosphere purification loop by a motor-driven blower. Flow from the cabin is through components in the following sequence: (1) coarse and fine elements of a particulate filter assembly, (2) a canister containing an activated charcoal filter for removal of gaseous contaminants, and lithium hydroxide for removal of CO_2, (3) a cabin loop blower, (4) a pressure suit boost blower, which is inactive in the normal operating mode, (5) a humidity control heat exchanger, which cools the cabin atmosphere and causes condensation of water vapor to liquid droplets, and (6) a motor-driven separator, which delivers nearly gas-free condensate to the water management system and nearly liquid-free atmosphere to the cabin.

The pressure suits provide backup capability for crew survival if cabin pressurization fails for any reason, or if there is accidental gross contamination of the cabin atmosphere. To convert the atmosphere purification loop to a pressure suit loop, the suit hoses are connected and the loop is isolated from the cabin by closing two valves. A suit boost blower is switched on to compensate for added pressure drop through the suits. A valve is opened to provide O_2, which is regulated to suit pressure. The suit loop is vented for sufficient time to purge most of the N_2 and enrich in O_2 to at least a normal partial pressure. The Apollo-type CO_2 absorbers have a capacity of 36 manhours. Each 18 hours the crew will remove a spent cartridge and replace it with a fresh one.

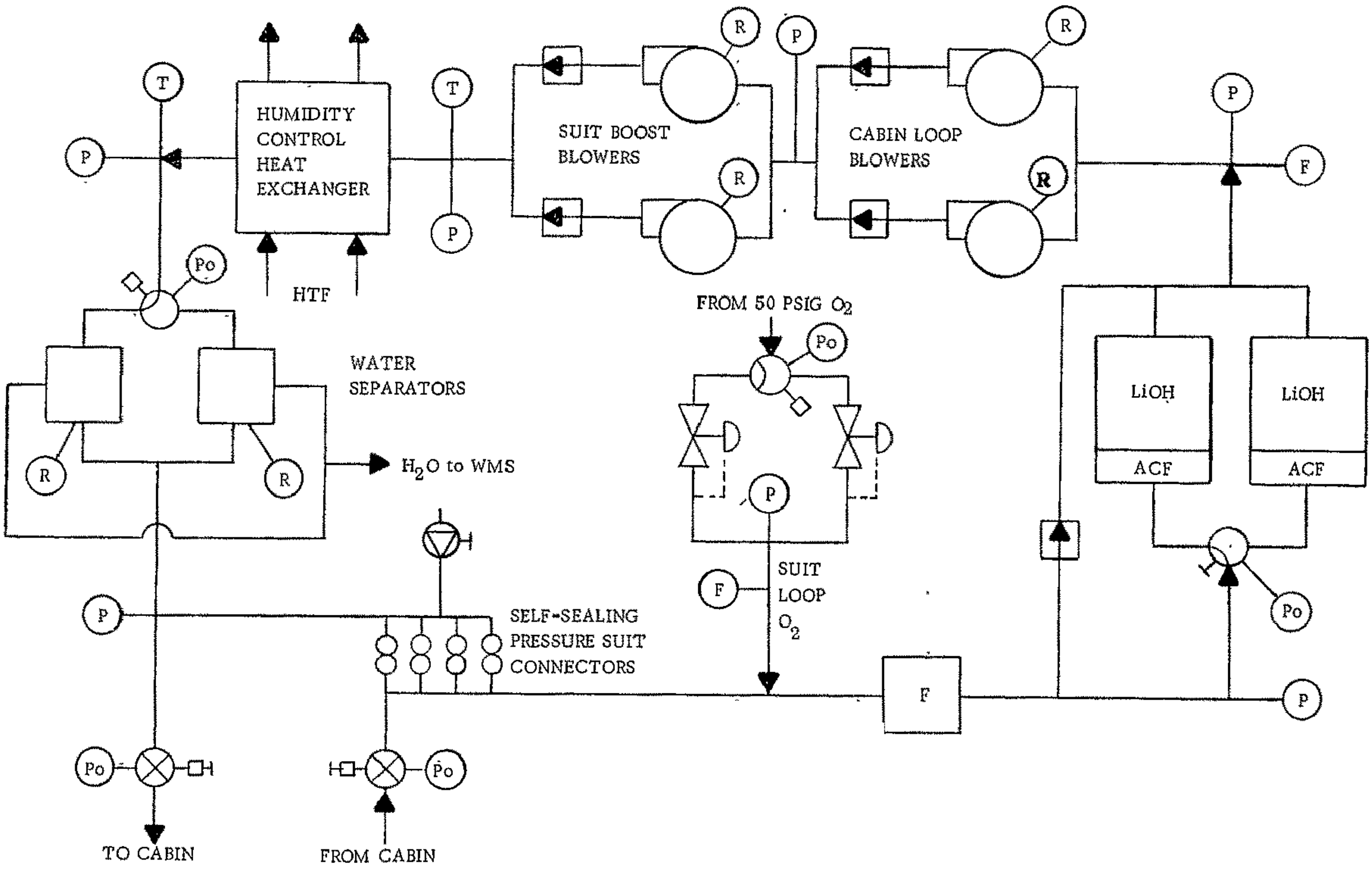

HUMIDITY CONTROL HEAT EXCHANGER
HTF
T
P
T
P
SUIT BOOST BLOWERS
R
R
P
CABIN LOOP BLOWERS
R
R
P
F
Po
WATER SEPARATORS
R
R
H2O to WMS
FROM 50 PSIG O2
Po
P
F
SUIT LOOP O2
LiOH
ACF
LiOH
ACF
Po
P
SELF-SEALING PRESSURE SUIT CONNECTORS
F
P
Po
Po
TO CABIN
FROM CABIN

Figure 5-2. Atmosphere Purification Loop

Table 5-13. Atmosphere Purification Loop Redundancies

LiOH and ACF	Redundant filter housing Expendable LiOH and ACF cartridge is replaced periodically
Cabin Loop Blower	Redundant blower Suit boost blower is backup
Suit Boost Blower	Redundant blower
Humidity Control Heat Exchanger	Redundant liquid passages in heat exchanger
Water Separators	Redundant water separator
Suit Loop Pressure Regulator	Redundant regulator

Included in the weights and volumes for the atmosphere purification subsystem, Table 5-14, are redundancies and fluid fill where applicable, and volume allowance for access.

Table 5-14. Atmosphere Purification Subsystem Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
Humidity Control Heat Exchanger (Dual)	23	1.2
Blowers	12	0.4
Water Separators	9	0.5
LiOH Canisters and Filters	65	3.0
Particulate Filters	5	1.5
Valves, Transducers, Regulators, Ducting, Supports	40	5.0
Total	154	11.6

5.4.3 WATER MANAGEMENT. Figure 5-3 shows the water management, including waste liquid storage and jettison. Backup and emergency modes are summarized in Table 5-15.

Water from the fuel cells is used for drinking and food and beverage preparation, with some of the excess stored for use in the thermal control sublimator for heat rejection. Humidity condensate, from the water separators of the atmosphere purification loop, is processed to achieve sterility and is used for personal hygiene and for flushing the urinal. Urine and used wash water are jettisoned. Storage is provided for water from all sources, which permits the backup uses and some choice of time and place for dumping wastes. Storage is in positive expulsion tanks which are pressurized with N_2 from the atmosphere stores.

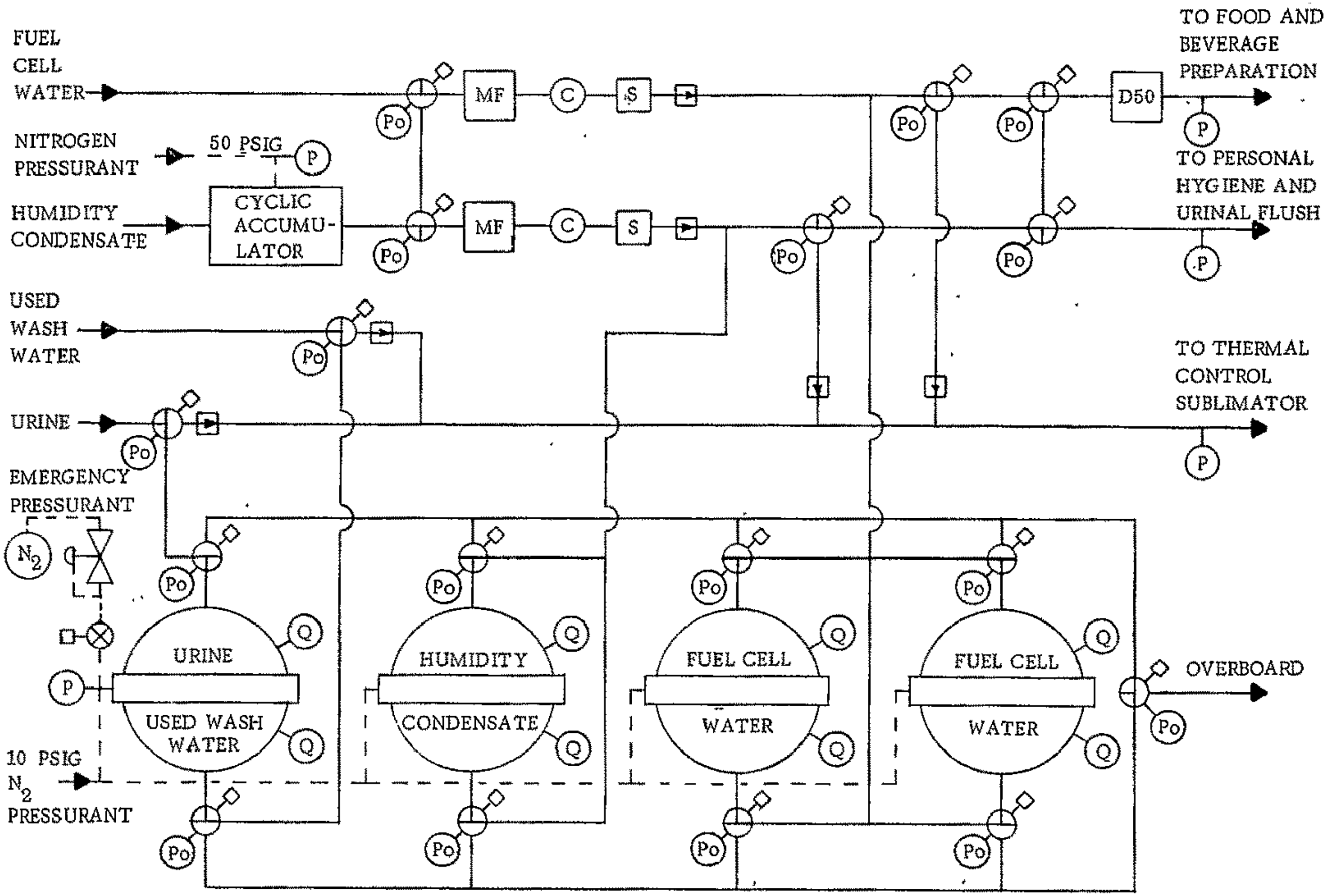

FUEL CELL WATER
NITROGEN PRESSURANT
50 PSIG
P
HUMIDITY CONDENSATE
CYCLIC ACCUMULATOR
USED WASH WATER
URINE
EMERGENCY PRESSURANT
N_2
10 PSIG N_2 PRESSURANT
Po
MF
C
S
D50
TO FOOD AND BEVERAGE PREPARATION
TO PERSONAL HYGIENE AND URINAL FLUSH
TO THERMAL CONTROL SUBLIMATOR
URINE
USED WASH WATER
HUMIDITY
CONDENSATE
FUEL CELL
WATER
FUEL CELL
WATER
Q
OVERBOARD

Figure 5-3. Water Management

Table 5-15. Water Management Subsystem Backup and Emergency Modes

Water Source	Primary Uses	Backup Uses	Emergency Use	Priority
Fuel Cell Water	Food and Beverage Heat Rejection	Personal Hygiene Urinal Flush.	Heat Rejection	1
Humidity Condensate	Personal Hygiene Urinal Flush	Food and Beverage	Heat Rejection	2
Used Wash Water	Dump Overboard		Heat Rejection	3
Urine	Dump Overboard		Heat Rejection	4

Processing the humidity condensate is through components in sequence as follows:

a. A cyclic accumulator pressurized with N_2 at 50 psig, which raises pressure for flow into the 10 psig water management system.

b. A multifiltration unit, consisting of a bacteria filter, an ion exchange column, and an activated charcoal filter.

c. A sterilizer, which adds a silver compound at sufficient concentration to arrest activity of microorganisms.

Fuel cell water is processed in a parallel set of components. If either set fails, humidity condensate is diverted to waste and fuel cell water is processed in the remaining set for continued use by the crew. A Dowex-50 ion exchange column removes the silver before water is used in food and beverages. Table 5-16 includes redundancies and volume allowance for installation and maintenance access.

Table 5-16. Water Management Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
Cyclic Accumulators	6	0.3
Multifiltration and Sterilizing Units	10	1.0
Emergency Pressurant and Regulator	5	0.5
Pressurized Storage Tanks	76	4.0
Valves, Transducers, Plumbing, Supports	33	3.0
Total	130	8.8

5.4.4 WASTE MANAGEMENT. Figure 5-4 illustrates the waste management subsystem. Backup modes are summarized in Table 5-17.

Table 5-17. Waste Management Backup Modes

Function	Component	First Backup	Second Backup
Feces Collection	Blower	Urine Collection Blower	Emergency Collector*
	Slinger	Reduced Capacity	Emergency Collector
	Others	Emergency Collector	Emergency Collector
Urine Collection	Blower	Feces Collection Blower	Emergency Urinal**
	Flush Accumulator	Emergency Urinal	Manual Flush
	Liquid-Gas Separator	Emergency Urinal	Plastic Bag

* Plastic glove type emergency feces collector.

**Personal hygiene waste water collector serves as an emergency urinal.

Urine is collected in a relief tube, into which cabin atmosphere is drawn to facilitate liquid capture in zero-g. Flow of the liquid-gas mixture is then directed into a motor-driven separator, which includes a pump for pressurized flow of the liquid into a storage tank of the water management subsystem. The separated gas flows through an activated charcoal filter for odor control, through the blower, and then returns to the cabin. The urinal is flushed with a metered quantity of water which contains a metered quantity of disinfectant. After each use the flush accumulator automatically refills with water and disinfectant.

The feces collector also utilizes entrainment in a flow of cabin atmosphere for transport of the waste into the collector. Cabin atmosphere return from the collector is through an activated charcoal filter and a blower back to the cabin. The feces impinge on a motor-driven "slinger," which spreads the fecal material on a plastic liner on the inner surface of the collector sphere. After use, the lid of the collector is closed and sealed. A selector valve exposes the sphere to space vacuum, which causes drying of the thin layer of fecal material. When thus dried, microbiological activity and decomposition are arrested. For the next use of the collector, the selector valve is moved to connect the sphere with the blower, at the same time shutting off the vacuum and allowing the sphere to repressurize with cabin atmosphere. Opening the lid actuates switches which start the slinger motor and the blower.

Estimated weight and volume of the unit are 56 lb and 3 ft^3. Additional volume is required for access to use the unit.

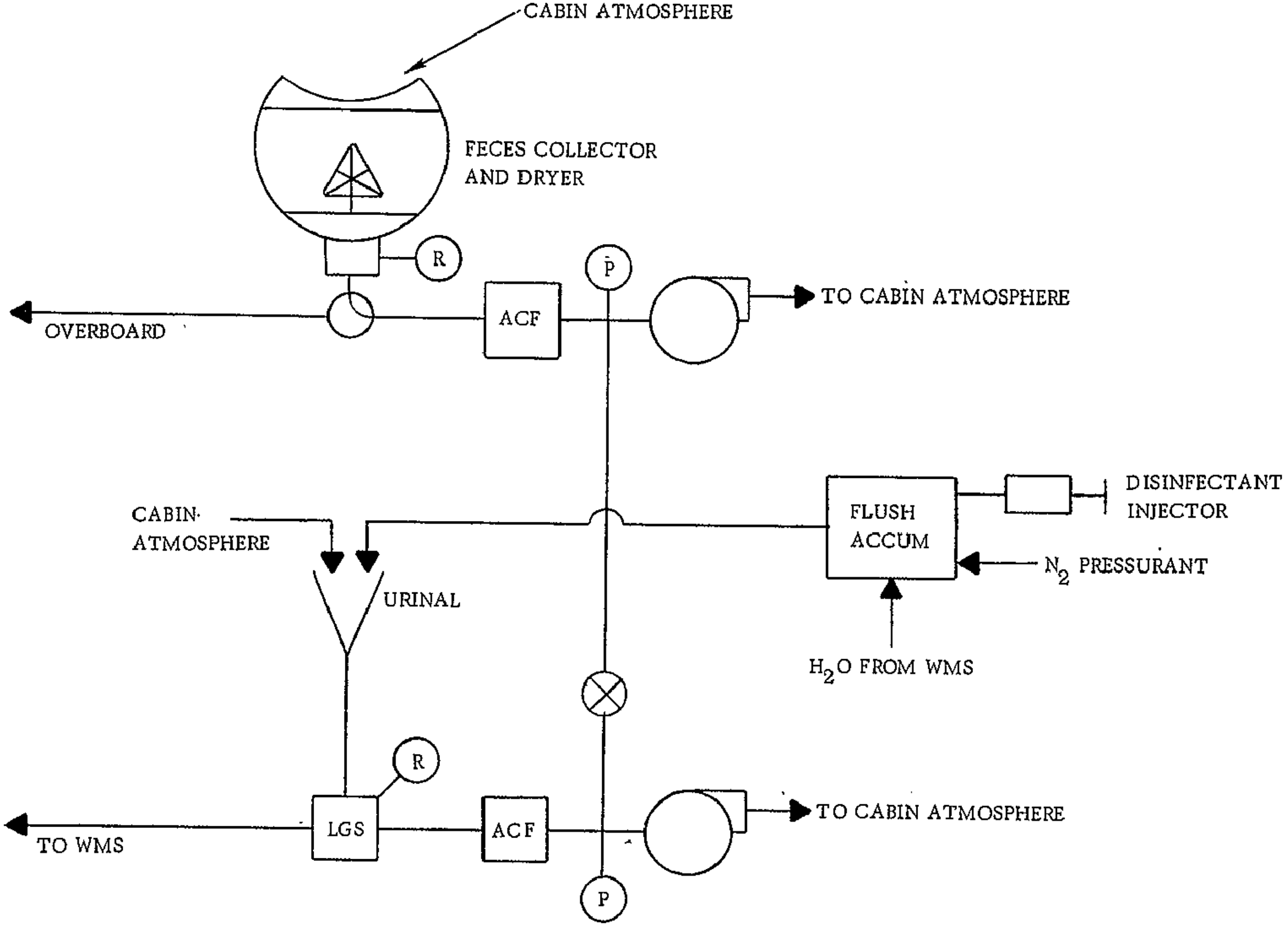

CABIN ATMOSPHERE
FECES COLLECTOR AND DRYER
R
P
ACF
TO CABIN ATMOSPHERE
OVERBOARD
DISINFECTANT INJECTOR
CABIN ATMOSPHERE
FLUSH ACCUM
N2 PRESSURANT
URINAL
H2O FROM WMS
R
LGS
ACF
TO CABIN ATMOSPHERE
TO WMS
P

Figure 5-4. Waste Management

5.4.5 FOOD MANAGEMENT. The hardware consists of a food storage container, a water heater chiller, and a water dispenser. The water heater and chiller are shown in the thermal control subsystem, Figure 5-6, and are included in weight estimates for thermal control. The food is considered to be of the Apollo type, with supplements selected for compatibility with zero-g, and is estimated at 2 lb/man-day. The food management components, including packaged food, have a weight of 32 lb and an installed volume of 2 ft^3.

5.4.6 PERSONAL HYGIENE. Figure 5-5 illustrates the facility for whole-body bathing by sponge. It can be used for any lesser degree of washing, such as for face and/or hands. The waste water collector is designed to serve as a backup to the urinal of the waste management subsystem.

The apparatus is designed for zero-g control of the wash water so as to minimize chance of cabin contamination. The user operates the apparatus and washes to the extent desired in a cycle as follows:

a. He opens the hinged lid on the end of the sponge wetter-squeezer, inserts the sponge, closes the lid, and presses a START button.

b. He waits a few seconds while the pneumatically-powered unit automatically cycles to (1) compress the sponge with the piston, expelling water and air, (2) release piston pressure on the sponge, (3) admit a metered quantity of water with detergent into the sponge, (4) vent the cylinder to admit cabin atmosphere and permit full return of the piston, and (4) unlatch the lid on the cylinder.

c. The user removes the sponge and washes a portion of his body.

d. He repeats the cycle as many times as desired for partial or whole-body bathing.

Whenever water is expelled by compressing the sponge, it flows into a waste water collector which is identical with the urinal. It has a motor-driven liquid/gas separator, with pump to transfer the liquid into a waste storage tank and a blower which returns entrained gas back to the cabin atmosphere. An activated charcoal filter adsorbs odors from the cabin atmosphere return.

The water for washing is from stored humidity condensate. It passes through a liquid-to-liquid heat exchanger where it is warmed by heat transport fluid of the thermal control subsystem. A liquid non-foaming detergent such as benzalkonium chloride (BAC) is metered into the wash water.

There are no redundancies in the personal hygiene subsystem. The backup consists of chemically-treated "dry wipes" which are used for cleaning hands and face only.

Table 5-18 gives the weights and volumes of this system.

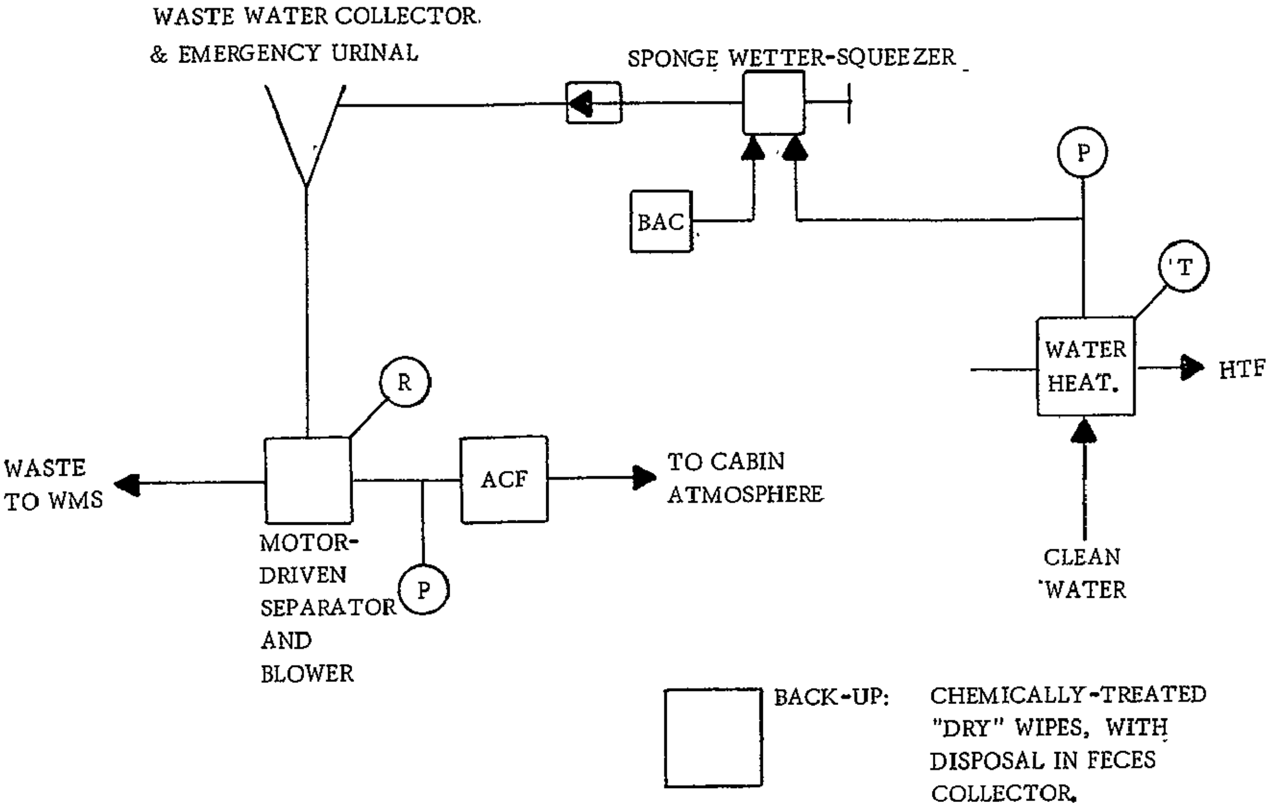

WASTE WATER COLLECTOR & EMERGENCY URINAL
SPONGE WETTER-SQUEEZER
P
BAC
T
WATER HEAT.
HTF
R
WASTE TO WMS
ACF
TO CABIN ATMOSPHERE
MOTOR-DRIVEN SEPARATOR AND BLOWER
P
CLEAN WATER
BACK-UP:
CHEMICALLY-TREATED "DRY" WIPES, WITH DISPOSAL IN FECES COLLECTOR.

Figure 5-5. Personal Hygiene

Table 5-18. Personal Hygiene Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
Water Heater, Controls, Plumbing	7	-
Sponge Wetter-Squeezer	5	-
Waste Water Collector, Blower, ACF	15	-
Total	27	1.2

5.4.7 THERMAL CONTROL. Figure 5-6 is a schematic of the thermal control subsystem. The redundancies and backups are summarized in Table 5-19. The thermal control subsystem has the functions of cabin atmosphere cooling, equipment cooling, heat transport, and heat rejection.

Although some heat may be transferred out through the cabin walls, most of the cabin atmosphere cooling load is shared between the humidity control heat exchanger and the cabin heat exchanger. The former operates in the atmosphere purification loop at temperatures low enough for moisture condensation and removal. The latter operates at temperatures low enough for cooling but high enough that moisture does not condense, and is independent of the atmosphere purification loop. Both transfer heat from the cabin atmosphere to a heat transport fluid (HTF). The humidity control heat exchanger

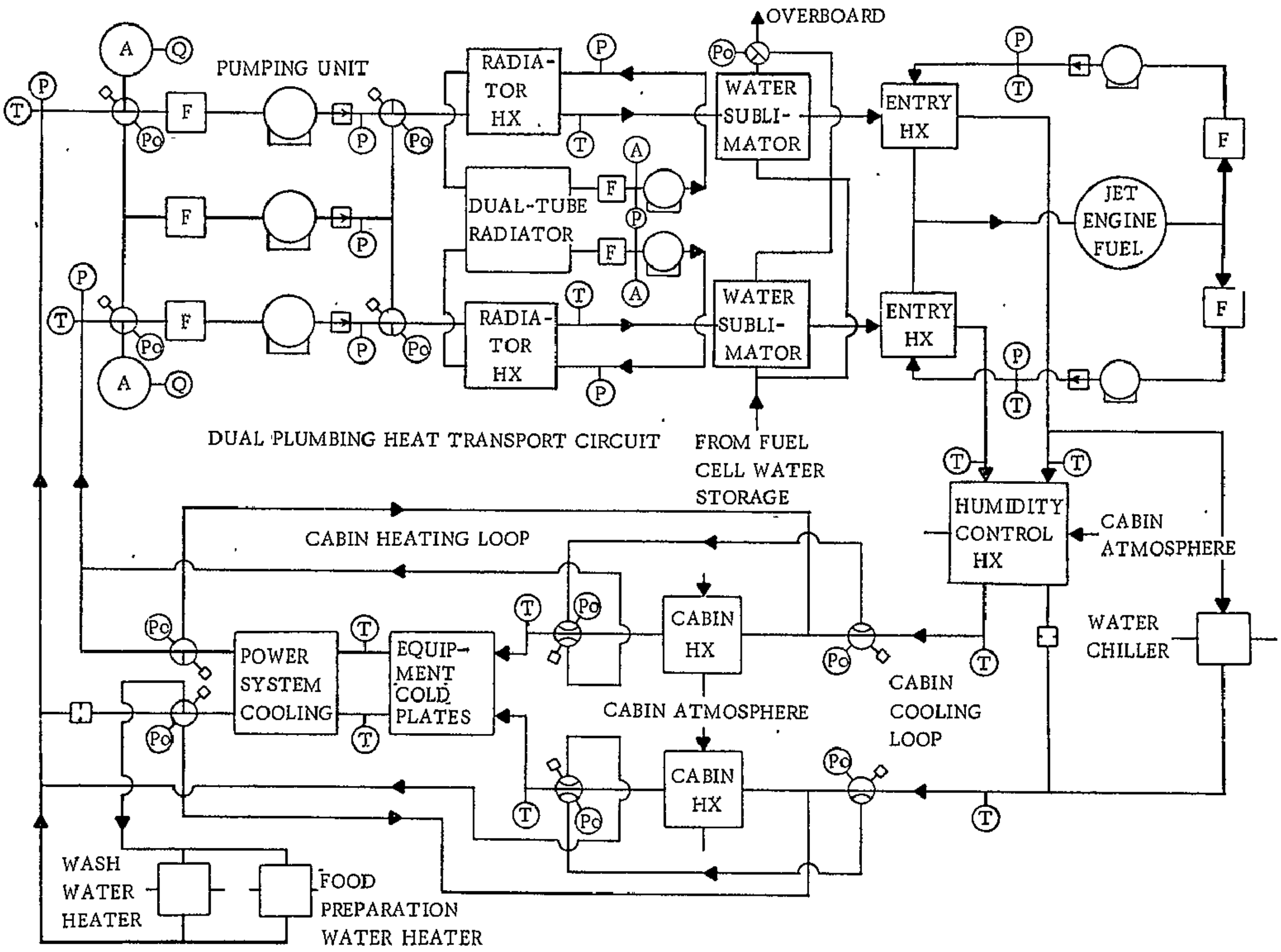

OVERBOARD
PUMPING UNIT
RADIATOR HX
WATER SUBLIMATOR
ENTRY HX
DUAL-TUBE RADIATOR
JET ENGINE FUEL
DUAL PLUMBING HEAT TRANSPORT CIRCUIT
FROM FUEL CELL WATER STORAGE
HUMIDITY CONTROL HX
CABIN ATMOSPHERE
CABIN HEATING LOOP
WATER CHILLER
POWER SYSTEM COOLING
EQUIPMENT COLD PLATES
CABIN HX
CABIN COOLING LOOP
WASH WATER HEATER
FOOD PREPARATION WATER HEATER
A
Q
P
T
Pc
F

Figure 5-6. Thermal Control System

has fixed flows which maintain relative humidity within the wide control band. The cabin heat exchanger has bypass variable flow in response to thermostat signals such that nearly constant temperature is maintained. The thermostat has an adjustable set point for selection by the crew as comfort may indicate.

Provision is made for operating the cabin heat exchanger in a heating mode. A selector valve diverts cold HTF around the heat exchanger and supplies it with hot HTF from the point of maximum temperature.

Table 5-19. Thermal Control Redundancies and Backups

Component	Redundancies	Backups
Radiator	Redundant tubing	Water sublimator Entry HX
Piping	100% redundancy except in functions of heating and cooling water for crew use.	
Pumps and Filters	Pump in redundant plumbing.	Additional redundant pump. Valves are controllable such that any one of the three pumps can operate in either plumbing circuit.
Humidity Control HX	Redundant liquid passages.	
Cabin HX	One redundant	Suit loop
Entry HX	One redundant	

Both the cabin atmosphere and liquid-cooled cold plates participate in cooling of electrical and electronic equipment. The estimated load division is shown in Table 5-6. Since the cabin atmosphere is cooled by the same heat transport fluid as the cold plates, all equipment heat contributes to temperature rise in the HTF. Components are arranged in a series-parallel network such that HTF flows and temperatures are consistent with component heat loads and temperature requirements. The chiller for crew drinking water is at the cold end of the circuit, and the heaters for personal hygiene and food preparation water are at the hot end.

The heat transport fluid is water with a corrosion inhibitor. It is filtered before entering the motor-driven pump, which circulates it at a rate consistent with the heat loads and temperature limits. A pump failure or loss of fluid causes automatic switching to the redundant circuit. An accumulator provides compensation for temperature-caused volume variations and has also a small fluid reserve for minor leaks.

In orbital flight the HTF is cooled in a liquid-to-liquid heat exchanger which transfers the heat into the radiator subsystem. The radiator fluid is Freon-21, selected for its viscosity characteristics at low temperatures. The radiator subsystem has the same redundancies in pumps, filters, and plumbing at the cabin heat transport circuit.

Heat rejection capacity of the radiator is supplemented by a water sublimator which uses fuel cell water. This use is for brief periods of peak heat loads or of unfavorable thermal radiation exposure for the radiator.

Another liquid-to-liquid heat exchanger is used during the period of entry and atmospheric flight. The cabin HTF transfers heat into the turbojet fuel as described in the trade studies of Section 5.2. This heat sink loop can be placed in operation at any time during flight. Within the considerable thermal capacity of the fuel, it can serve as a backup to the radiator and water sublimator.

The radiator concept is of panels in the upper and lower surfaces of the wings as evolved in the trade studies of Section 5.2. The wing skin is designed of a thickness to serve as radiator fin material. Based on the total heat load stated in Table 5-6 the estimated area requirement is 452 ft^2 on each the upper and lower surfaces of the wings.

The radiator weight shown in Table 5-20 is the estimated increase in wing weight to incorporate radiators, where optimization is based on both thermal and structural parameters. All redundancies are included.

Table 5-20. Thermal Control Weights and Volumes

Item	Weight (lb)	Volume (ft^3)
Radiator Panels	452	
Radiator Fluid Fill	90	
Radiator Controls	35	1
Radiator Plumbing	20	1
Radiator Pumping Unit	25	1
HTF to F-21 H-X	20	1
Entry HX, Pumping Unit, Fluid Fill	58	2
Water Sublimators	26	1
Cabin Air HX and Blowers	60	5
HTF Pumping Unit	25	1
Water Heater and Chiller	10	1
Valves, Transducers, Controls, Plumbing	19	2
Heat Transport Fluid Fill	60	
Total	900	16

5.4.8 ORBITER EC/LSS WEIGHT AND VOLUME SUMMARY. Table 5-21 summarizes orbiter EC/LSS weights and volumes, the latter reflecting selection of equipment locations within and outside the cabin. The outside location is an equipment bay. These are estimates based on criteria that (1) certain equipment is directly used by the crew and must be within the cabin, (2) to the extent feasible equipment not required in the cabin should be in the equipment bay, thus minimizing use of cabin volume, and (3) penalties in plumbing complexity and weight are to be avoided.

5.5 DETAIL DESIGN DATA, BOOSTER

Oxygen storage is described in Paragraph 5.2.2.3. The only other required EC/LSS function is thermal control for equipment and for the cabin atmosphere. Humidity control of the cabin atmosphere is considered to be a thermal function, consisting of cooling, condensing, and gravity drain of the condensate. The estimated electrical power requirement is 150 watts. Figure 5-7 is a schematic of the thermal control subsystem. Table 5-22 summarizes the weights and volumes.

Table 5-21. Orbiter EC/LSS Weight and Volume Summary

	Weights (lb)			Volumes (ft^3)		
	Expendable	Non-Expendable	Total	Inside Cabin	Outside Cabin	Total
Atmospheric Stores and Pressurization Control	83	113	196	1.8	9.2	11.0
Atmosphere Purification	47	107	154	7.2	4.4	11.6
Water Management		130	130	1.0	7.8	8.8
Waste Management		56	56	3.0		2.0
Food Management	28	4	32	2.0		3.0
Personal Hygiene		27	27	1.2		1.2
Thermal Control		900	900	5.0	11.0	16.0
Total	158	1337	1495	21.2	32.4	53.6

5.6 EXTENDED DURATION

The primary expendables are oxygen, CO_2 absorber, and food. There are also expendables in very small quantities for personal hygiene, waste management, and water management. Another consideration is that the waste collector must provide for the greater volume accumulated in extended durations.

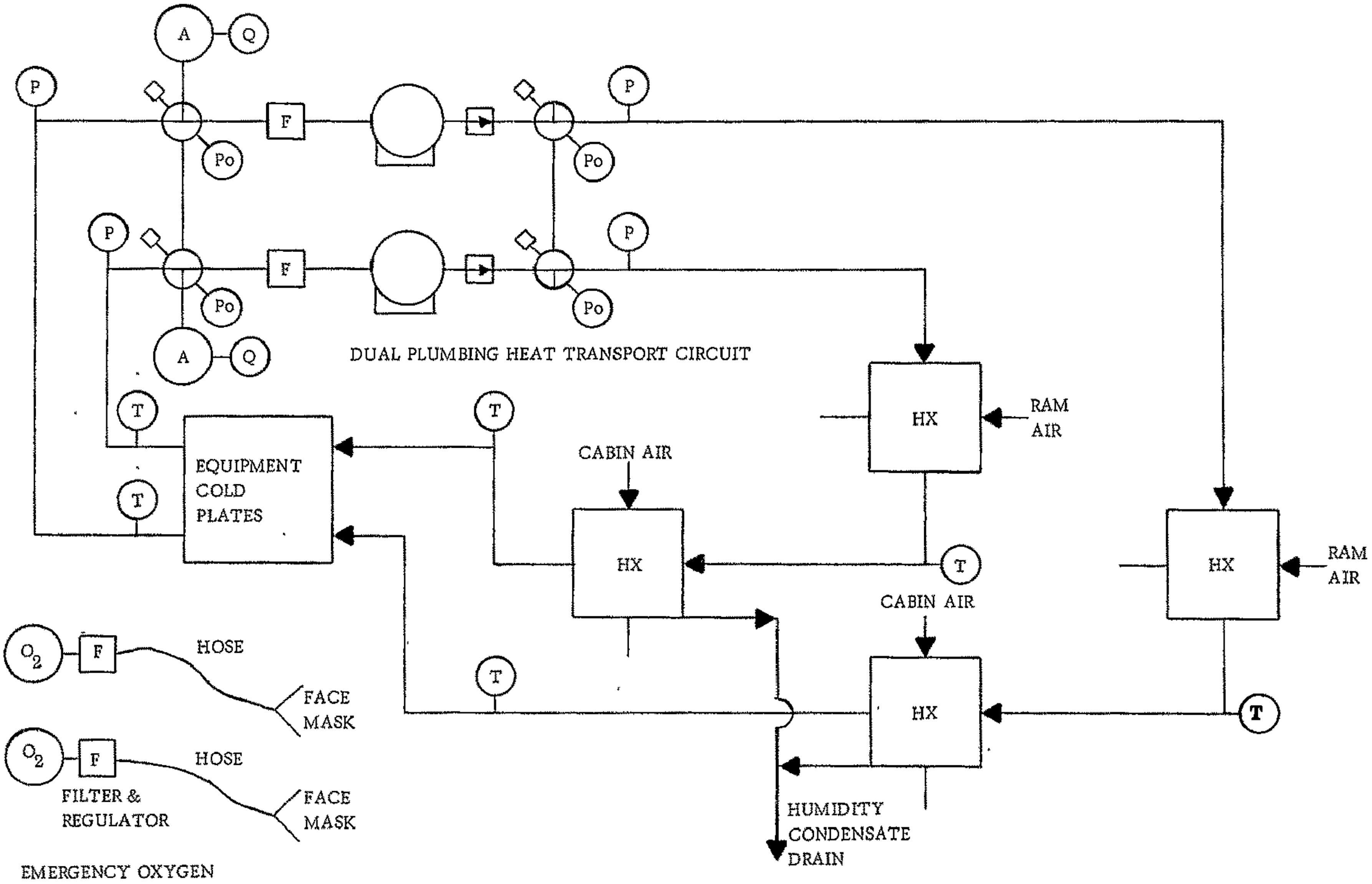

A
Q
P
F
Po
P
Po
P
F
Po
P
Po
A
Q
DUAL PLUMBING HEAT TRANSPORT CIRCUIT
T
T
EQUIPMENT COLD PLATES
T
HX
RAM AIR
CABIN AIR
HX
T
HX
RAM AIR
CABIN AIR
O_2
F
HOSE
FACE MASK
T
HX
T
O_2
F
HOSE
FILTER & REGULATOR
FACE MASK
HUMIDITY CONDENSATE DRAIN
EMERGENCY OXYGEN

Figure 5-7. Booster Thermal Control

5.6.1 OXYGEN STORAGE FOR EXTENDED DURATION. The O_2 for normal seven-day missions is stored in the equipment bay, with a supply line to the cabin. Storing an additional quantity in the payload bay will have the same kind of interface, but with a longer line. Quantity requirements are given in Paragraph 5.1 and vessel weight data are described in the trade studies of Paragraph 5.2.

Table 5-22. Booster EC/LSS Weight and Volume Summary

		Cabin Volume (ft^3)		
Item	Weight (lb)	Inside	Outside	Total
Cabin Air Heat Exchangers and Blowers	42	2		2
Ram Air Heat Exchangers	11		1	1
Heat Transport Fluid Pumping Unit	25		1	1
Heat Transport Fluid Fill	30			
Gaseous Oxygen System	18			
Valves, Transducers, Controls, Plumbing	10	1	–	1
Total	136	3	2	5

5.6.2 CO_2 ABSORBER STORAGE FOR EXTENDED DURATION. The LiOH-ACF cartridges are normally stored in the cabin so that the crew can readily make the scheduled replacements of spent cartridges. Storage in the payload bay would impose a requirement to transfer spent cartridges from the cabin to the payload bay, and fresh cartridges from the payload bay to the cabin.

Transfers in shirtsleeve attire will require that the payload bay and a passage to it shall be pressurized. If not, the crewman must wear a pressure suit and portable life support system, and must pass through an air lock. If the payload bay and passage are to be pressurized for other reasons, the shirtsleeve exchange of CO_2 absorber supplies and wastes is a reasonable approach. If not, storage in the cabin is recommended.

5.6.3 FOOD FOR EXTENDED DURATION. Storage in the cabin is recommended unless pressurization of the payload bay and passage is planned for other reasons.

5.6.4 WASTE MANAGEMENT FOR EXTENDED DURATION. The waste collector sphere described in Paragraph 5.4.4 is of a volume corresponding to the 14 man-day mission. One or more replacement spheres could be stored in the payload bay, subject to the same transfer problems described for the CO_2 absorber. An alternate approach is to store the replacement spheres in the cabin.

The collector sphere can be of larger volume for increased capacity. The increment is about 0.025 ft^3/day or 40 days/ft^3 for a 2-man crew. A reasonable size limit for convenient handling is about 1.8 ft^3 and 120 man-day capacity, based on $V = 0.3$ ft^3 + 0.0125 ft^3/man-day. For extended durations the larger collection sphere and longer change interval is recommended over the seven-day size and change interval.

5.6.5 PARAMETRIC DATA FOR EXTENDED DURATION. EC/LSS weights and volumes were determined parametrically with duration for a vehicle configuration having separate crew and passenger compartments. The crew cabin is occupied by two crewmen and the flight equipment described in Section 5.7.5. The passenger cabin has a separate EC/LSS sized for 12 men but is occupied by only 10 men. Therefore, the total vehicle capacity is for 14 men and expendables are for 12 men.

It was assumed that cabin atmosphere leakage is directly proportional to cabin volume, which is directly proportional to the man capacity of the cabin. This permits leakage to be treated parametrically with other expendables having use rates per man-day. The crew cabin will have excess water, which will more than supply the deficit in the passenger cabin. It is assumed that plumbing will permit this water transfer. The other quantities which are variable with duration are shown in Table 5-23.

Table 5-23. EC/LSS Expendables Rates

Item		Weight (lb/man-day)	Volume (ft^3/man-day)
O_2 Consumption	1.7		
O_2 Leakage	0.3		
Subtotal	2.0		
Redundancy @ 25%	0.5		
Subtotal	2.5	2.5	
O_2 Vessel Weight	2.5(0.205)	0.51	
O_2 Vessel Volume	2.5(0.023)		0.057
N_2 Leakage		0.68	
N_2 Vessel Weight	0.68(0.236)	0.16	
N_2 Vessel Volume	0.68(0.032)		0.022
LiOH/ACF		3.13	0.133
Food, Packaged		2.00	0.100
Total		8.98	0.312

The foregoing rates, rounded to 9 lb/man-day and 0.3 ft^3/man-day, were used to obtain the weight and volume summary shown in Table 5-24. The electrical power requirement does not vary with mission duration and is estimated to average 1570 watts.

Table 5-24. Weight and Volume Summary, 30-Day Mission

	Weight (lb)	Volume (ft^3)
System Hardware†		
2-Man Crew Cabin	1308	48
12-Man Passenger Cabin	1641	64
Subtotal	2949	112
Expendables††		
7 Days, 12 Men	756	25
Subtotal (7 days)	3705	137
Expendables††		
23 Days, 12 Men	2484	83
Total (30 days)	6189	220

† Excludes containers for expendables

†† Includes containers for expendables

For the two-cabin, 14-man capacity configuration described, with 12-man occupancy, the weight and volume equations are:

$$W = 2949 + 108\,D \text{ (lb)}$$

$$V = 112 + 3.6\,D \text{ (ft}^3\text{)}$$

where

D = number of days mission duration.

5.7 FAILURE MODES, DETECTION AND ACTION

This paragraph describes the Environmental Control/Life Support System (EC/LSS) in terms of composition, normal operation, failure modes, failure sensing, and the redundancies and backups for restoration of function. The rationale for redundancies and backups is to make provision for failure in functions such that a first failure does not require mission abort (fail operational) and a second failure does not prevent safe abort (fail safe). In some cases there are several levels of backup capability.

5.7.1 ATMOSPHERE SUPPLY AND PRESSURIZATION CONTROL. Refer to Figure 5-1 and Table 5-11 for details.

5.7.1.1 First Failures, Atmosphere Supply and Pressurization Control Insulation Failure in One Vessel. Heat influx will cause rapid pressure rise and fluid loss by venting through relief valves. Loss is detected by quantity gaging in the defective vessel. Redundancy is such that sufficient O_2 and N_2 remains in other vessels to complete the mission. Capability to completely repressurize the cabin may be lost.

Leakage of One Vessel. Loss is detected by quantity gaging. Redundancy is such that sufficient O_2 and N_2 remains in other vessels to complete the mission. Capability to completely repressurize the cabin may be lost.

Heater Failure. Failure is indicated by measurement showing zero current in the heater. Each vessel contains a redundant heater, which is switched on to restore function.

Flow Restrictor Blockage. Failure is detected by high Δp across the restrictor. Flow is restored by opening a solenoid-operated valve in the restrictor bypass line. This valve is also opened to achieve high flow for rapid cabin repressurization under manual control.

Filter Blockage. Failure is detected by high Δp across the filter. Flow is restored by opening the solenoid-operated valve to a redundant filter.

Failure of Pressure Regulators. Failure open is detected by abnormally high pressure downstream of the regulator, and/or by opening of the relief valve which prevents over-pressure damage to downstream components. Failure closed is detected by abnormally low downstream pressure. Function is restored by actuating a selector valve which switches flow to a redundant regulator.

Two-Gas Pressure Control. Failure is detected by periodic analysis of the cabin atmosphere for O_2 partial pressure and by reading cabin total pressure. An overpressure failure is indicated by opening of the cabin pressure relief valve. Function is restored by manual switching to redundant sensing and control elements in the unit.

Cabin Pressure Relief Valve. Failure closed will cause the function to be assumed by a redundant relief valve. No switching is required. Failure open is detected by sound or by abnormally high flow from atmosphere stores. This failure is corrected by manual override.

Cabin Leakage. High rate loss of cabin atmosphere is detected by sound or by drop in cabin total pressure due to limited inflow. Low rate loss may be detectable only by periodic quantity gaging of atmosphere stores which will show abnormally high use rate. High rate loss may be stopped or reduced by locating the defect and applying a sealant. Atmosphere stores have sufficient reserve to sustain a low rate loss without mission abort.

5.7.1.2 Second Failures. If a second O_2 vessel fails in a mode causing loss of supercritical fluid, sufficient O_2 is available in the cabin atmosphere and/or in other vessels to sustain the crew for several hours. If the failure is in the second N_2 vessel, sufficient N_2 will remain in the cabin that no additional inflow is required. The N_2 partial pressure will decrease slowly due to leakage. Failure of the second heater in a vessel results in partial failure of the vessel. Gas will be delivered at a reduced rate as limited by heat leak through the insulation. Redundancy is such that sufficient O_2 and N_2 can be delivered from other vessels to complete the mission.

If a redundant filter fails by blockage, flow is restored by opening a solenoid-operated bypass valve. If a redundant O_2 pressure regulator fails, function is restored by switching flow through another redundant regulator. If the redundant N_2 regulator fails closed, N_2 partial pressure in the cabin is allowed to diminish slowly due to leakage. This does not create a hazard to the crew. If the regulator fails open, N_2 flow is stopped by closing three solenoid-operated valves upstream of the filters. If the redundant elements of the two-gas pressure control fail, manual overrides are used to periodically admit O_2 and N_2 to the cabin to restore correct partial pressures.

If a combination of failures is such that function of the primary N_2 source cannot be restored, the water management subsystem is supplied pressurant from a backup source. If serious leakage or a combination of failures is such that cabin pressurization cannot be maintained, the crew dons pressure suits which are connected into a suit loop for O_2 supply, atmosphere purification, humidity control, and temperature control. The same backup applies to gross atmospheric contamination which may require intentional venting of the cabin without capability to repressurize.

5.7.2 ATMOSPHERE PURIFICATION. Figure 5-2 shows the major components of the subsystem with redundancies summarized in Table 5-13.

5.7.2.1 First Failures, Atmosphere Purification Loop Particulate Filter. The particulate filter receives almost no loading in normal operation. A channeling or by-pass type failure reduces effectiveness and may be detected by an abnormally low Δp across the filter. A blockage or contamination type failure will require removal of the failed filter element, and may be detected by an abnormally high Δp across the filter.

ACF/LiOH Filter Assembly. The cartridge within the filter assembly is expendable and is replaced on an established schedule. Depletion of the LiOH may be detected by gas analysis showing increased CO_2 in the cabin atmosphere. Depleted ACF capacity may be detected by odor. Depletion will not occur in normal operation if the cartridges are replaced on schedule. To restore function after any type failure, a redundant filter assembly is switched on the line by means of a selector valve.

Cabin Loop Blower. Failure is detected by abnormally low (or zero) rotational speed of the impeller, by sound, or by failed Δp across the blower. Function is restored by switching on the redundant blower.

Humidity Control Heat Exchanger. Liquid side redundancy is discussed in Paragraph 5.7.7, Thermal Control. The gas side of the heat exchanger is assumed to be equivalent to ducting in which no uncorrectible failures will occur.

Water Separators. Failure is detected by sensing abnormally low (or zero) rotational speed or by abnormal pressures in the gas and liquid streams. Function is restored by switching on the redundant water separator.

5.7.2.2 Second Failures. If the second cabin loop blower fails, function is restored by switching on a suit boost blower. One ACF/LiOH cartridge is carried in excess of scheduled replacement needs. If there is a second failure in the water separators or humidity control heat exchanger, some backup exists in the capability of the cabin atmosphere to take on added moisture up to a point of near saturation. Additional backup is attainable by manually venting moisture-laden cabin atmosphere and making replacement with dry gases from storage.

In addition to other redundancies of the atmosphere purification loop, there is a redundant suit boost blower and a redundant O_2 pressure regulator. Failure of a blower will be immediately apparent to the crew and will also show as reduced Δp across all components. Correction is made by switching on the redundant blower. Failure of the O_2 pressure regulator will show on the suit pressure gage and is corrected by switching flow through the redundant regulator.

5.7.3 WATER MANAGEMENT. Figure 5-3 shows the water management scheme with backup and emergency modes summarized in Table 5-15.

5.7.3.1 First Failures, Pressurization. Gross failure in pressurization is detected by pressure measurement in the pressurant lines and in the water lines. If failure is in the N_2 pressurant supply, there are redundant components to restore function. (See Section 5.7.1.) Pressurant leakage is detected by N_2 quantity gaging which will show abnormally high use rate. A low rate leak can be sustained for the mission duration. A high rate leak requires that the failed portion of the plumbing be isolated by valves.

Humidity Condensate. Normal use of humidity condensate is for personal hygiene and urinal flush, both of which can be terminated without requiring mission abort. However, selector valves permit use of fuel cell water for these purposes. Failure of the multifiltration unit purification function may be detected by a conductivity sensor, by sampling and analysis, or by odor, but this kind of failure does not necessarily preclude use of the water. A failure of the cyclic accumulator pressurization function, or a blockage downstream, causes water accumulation in the water separators. This is detected by discharge of water drops into the cabin and by abnormal Δp across the separator. If one of these failures occurs, the humidity condensate is collected in a plastic bag which is periodically emptied manually into either the urinal or the waste wash water collector. A leakage type failure is detected visually or by storage tank gaging which shows abnormally low quantity. Water which leaks may evaporate into the cabin atmosphere and be returned as humidity condensate, or it may be picked up manually in a sponge from which it is discharged into the personal hygiene waste water. (See Paragraph 5.7.1.)

Fuel Cell Water. Normal use of fuel cell water is for drinking and for food and beverage preparation. These uses take only about one-third of the water produced, the remainder being stored. Although this storage is primarily for potential use in the thermal control sublimators as a backup mode of heat rejection, the water can be used for any other purpose. If there is complete failure of the fuel cell water supply and the stored water has been used, needs for drinking and food preparation can be partially met by using processed humidity condensate.

5.7.3.2 Second Failures. There are several levels of backup capability for failures beyond the first. For emergency pressurization of the water system there is a separate source of gaseous N_2. If this also fails, the O_2 will be used as a pressurant. Isolation of the various water sources precludes loss of all water in a second plumbing or component failure. If there is a heat rejection emergency, selector valves permit use of any water on board for this purpose. After using all available fuel cell water, the priorities in emergency diversion of other water to the thermal control sublimator will be 1) humidity condensate, 2) used wash water, and 3) urine with urinal flush water.

5.7.3.3 Performance Monitoring, Servicing, and Selector Valve Checkout. Performance of the multifiltration units is monitored during flight by periodically measuring conductivity of the product water. Failure in purification is indicated by high conductivity. If this occurs in water for food and beverage preparation, valves are actuated to select the alternate source. If it occurs in water for personal hygiene and urinal flush, these uses can continue.

As soon as the vehicle is committed to reentry, all fresh and waste water on board is jettisoned. Valves are actuated to isolate the water management system and all lines and components are evacuated to space. After landing, the system is flushed with water and disinfectant and purged dry with clean nitrogen. Adequate flushing will require valve actuation, which will serve as a valve function checkout. Valve actuation can also be checked at any time while the system is dry.

5.7.4 WASTE MANAGEMENT. Figure 5-4 illustrates the waste management subsystem. Backup modes are summarized in Table 5-17.

5.7.4.1 First Failures. Blower failure may be detected by sound, by sensing rotational speed of the impeller, or by sensing pressure rise across the blower. If either blower fails, the other blower can be turned on to cause the flow of cabin atmosphere required for zero-g waste collection. Selector valves and electrical switches are used for transfer of the function. Failure of the slinger or liquid-gas separator will be detected by sound or by sensing rotational speed. If the liquid-gas separator fails, the waste water collector of the personal hygiene subsystem is used as an emergency urinal. If the slinger fails, feces collection can continue but at less capacity of the container. Also, the fecal material will not dry enough to inhibit bacterial activity. If the flush accumulator fails, the urinal can be flushed manually by discharging water into it from a plastic bag or from a sponge. The failure would be observed visually.

5.7.4.2 Second Failures. If the second blower fails, the waste water collector of the personal hygiene subsystem is used as an emergency urinal. The emergency feces collector is a "defecation glove." After use the collector and contents are stored in the feces collection sphere. Partial drying may be attained if vacuum integrity of the sphere and functioning of the selector valve are intact.

5.7.5 FOOD MANAGEMENT. The hardware consists of a food storage container, a water heater, a water chiller, and a water dispenser. The water heater and chiller are shown in the thermal control subsystem schematic, Figure 5-6. The food is considered to be of the Apollo type, with supplements selected for compatibility with zero-g. If there are failures in water heating or chilling, the water is consumed at the temperature available. An emergency water dispenser is provided.

5.7.6 PERSONAL HYGIENE. The system is diagrammed in Figure 5-5.

5.7.6.1 Failures. Whole body bathing cannot be accomplished when the personal hygiene subsystem fails. A backup mode available for cleansing the hands and face consists of chemically treated wipes. Disposal of the wipes is into the feces collector. Another mode is available if the supply and control of water and detergent are intact. In this case, small, untreated disposable wipes are wetted by manually controlling water flow into the cylinder, and are then used for cleaning the hands and face. Drying can be done with another untreated, disposable wipe. Disposal is also into the feces collector.

5.7.7 THERMAL CONTROL. Figure 5-6 is a schematic of the thermal control subsystem. The redundancies and backups are summarized in Figure 5-19.

5.7.7.1 First Failures. The thermal control subsystem has 100% redundancy in plumbing and in all components except the water coolers and heaters for personal hygiene and for food and beverage preparation. Failure of these components would not require abort. The 100% redundancy amounts to two independent heat transport circuits, two independent radiator heat rejection loops, and two independent fuel heat sink loops.

Each of these elements can handle the full thermal load. The failure of any pump or loss of fluid in any loop causes automatic switching to the redundant loop. Fluid loss failures are detected by accumulator quantity sensors. Pump failures are detected by measurement of Δp.

5.7.7.2 Second Failures. Triple redundancy is provided in the heat transport fluid pumping unit. By actuating electrical switches and selector valves, any of the three pumps can be used in either heat transport circuit. Pump failure is detected by Δp measurement.

The heat rejection function has several levels of backup. Each of the redundant radiator loops is backed up by a water sublimator and by a jet fuel heat sink loop. The priority in employing the backups is to use the water sublimator until the water supply is exhausted or until the vehicle descends into the atmosphere. Thereafter the jet fuel heat sink loop is activated by switching on a pump. If all heat rejection fails, the ultimate backup is in the thermal inertia of the cabin atmosphere, the heat transport fluid, and all the thermal control hardware. This backup has an estimated capacity for 15 minutes of operation at degraded thermal performance.

5.8 REFERENCES

5-1 Hamilton Standard Division of United Aircraft, The Porous Plate Sublimator for Cooling Applications in Space, PDB 6315, October 18, 1963.

5-2 Hamilton Standard Division of United Aircraft, Environmental Control System Component Data, Revised 7-1-66.

5-3 Convair Division of General Dynamics, Study of Subsystem Module Preliminary Definition, Final Report, Volume VII, "Radiators and Thermal Analysis" GDC-DAB67-003, Contract NAS 9-6796, October 1967.

5-4 Russ, E. J., King, C.D., and Drake, G. L., Preliminary Design Studies on Environmental Control/Life Support Systems for Maneuverable Spacecraft, General Dynamics, Convair division, Report GDC-ERR-1270, December 1968.

5-5 Convair Division of General Dynamics, Study of Basic Subsystem Module Preliminary Definition, Final Report, Volume VI, "Environmental Control and Life Support", GDC-DAB67-003, Contract NAS 9-6796, October 1967.

5-6 Hamilton Standard Division of United Aircraft, Final Report, Basic Subsystem Module, SP 67128, Prepared for Convair division. General Dynamics Corporation, August 21, 1967.

5-7 Convair Division of General Dynamics, Life Support System Comparison, Prototype Versus Flight, Life Support System for Space Flights of Extended Time Periods, Report No. 64-26235, Contract NAS 1-2934, 22 November 1965.

5-8 Fairchild Hiller Corporation, Space and Electronic Systems Division, Spacecraft Temperature Control Thermal Louvers.

SECTION 6

HYDRAULIC POWER GENERATION AND DISTRIBUTION

6.1 REQUIREMENTS

6.1.1 FUNCTIONS. The basic function of the hydraulic system is to provide fluid power for operation of the elevons, ruddervators, wing deployment, turbojet deployment, wing spoilers, wing flaps, landing gear and other miscellaneous work functions. (Electrical or pneumatic actuation is used for work functions such as cargo hatch and stage separation.)

6.1.2 POWER REQUIREMENTS. The magnitude of hydraulic system power is required in order to establish subsystem weight and to define the auxiliary power unit. Major subsystem demands are dictated by the requirement for aerodynamic control. The hinge moments and deflection rates for the elevons and ruddervators were established for early FR-4 configurations. (See Section 4.) The hydraulic pump shaft power is presented in Figure 6-1 (FR-4 Booster), Figure 6-2 (FR-4 Orbiter), Figure 6-3 (FR-3 Booster) and Figure 6-4 (FR-3 Orbiter). This power requirement is conservative; it is anticipated that further refinement of the control surface configuration will result in some reduction. In addition, the entry trajectory is a significant influence in hinge moment determination. Potential revisions to the trajectory and to estimated control surface deflection requirements indicate that a reduction of the power required for aerodynamic control is probable.

6.2 SYSTEM DESCRIPTION

Hydraulic power was selected for actuation of the various work functions. This decision was predicated on development status, response characteristics, and stiffness compared to competing electromechanical and pneumatic actuation systems. In addition, the hydraulic fluid can be used as a heat transfer medium to control the temperature of critical components, e.g., actuator rod-end bearings and dynamic seals.

Three independent hydraulic actuation circuits are provided as shown in Figure 6-5 and 6-6. All three circuits operate simultaneously to provide for actuation of essential work functions. These include elevons, ruddervators, wing doors, wing deployment, wing spoilers, turbojet doors, and turbojet deployment.

The utility work functions include wing flaps, main landing gear doors, nose landing gear, main wheel brakes, and nose wheel steering. These work functions are supplied by hydraulic circuits 2 and 3. Isolation valves are used to separate the utility

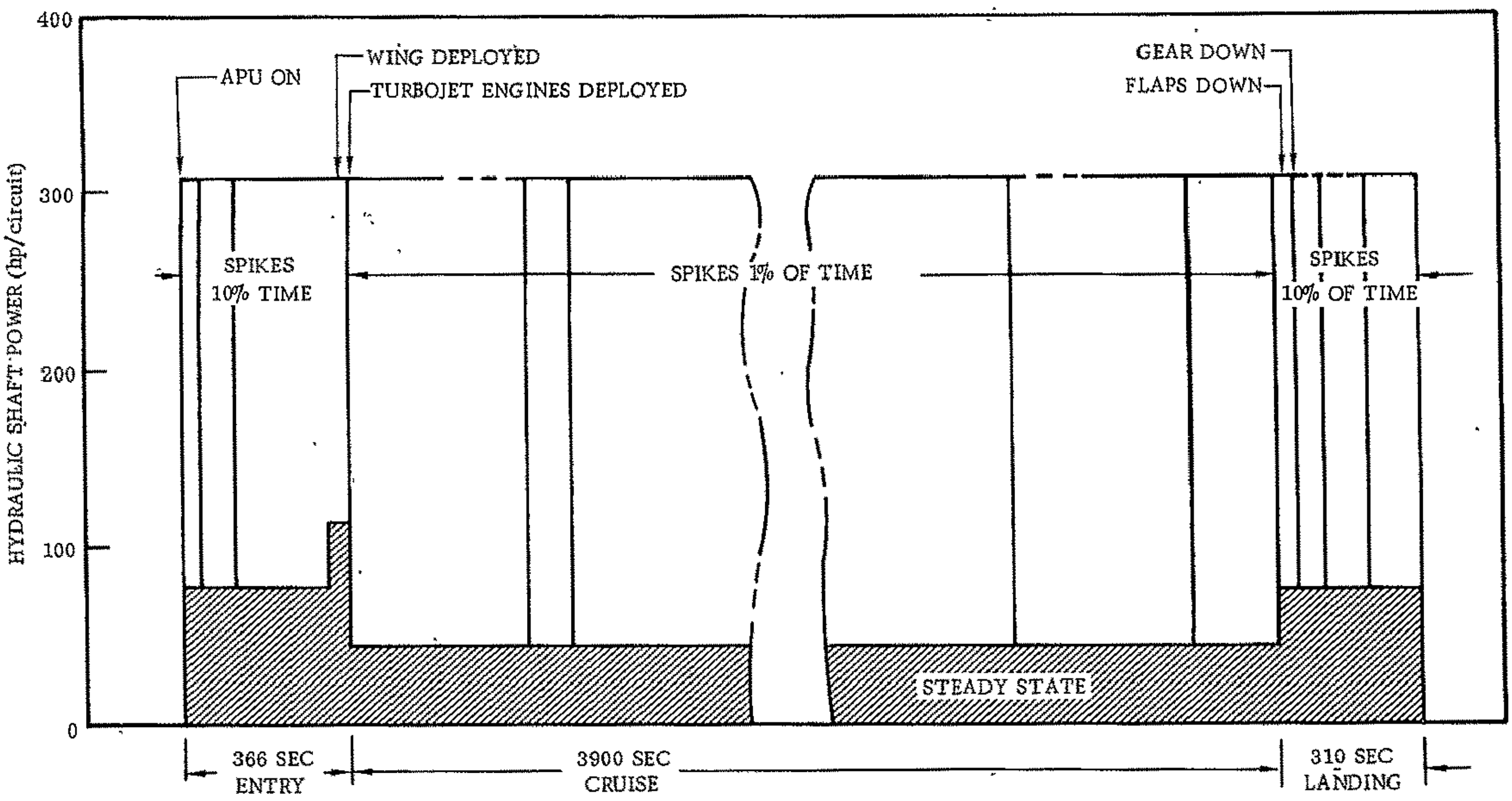


Figure 6-1. FR-4 Booster Hydraulic Power Profile

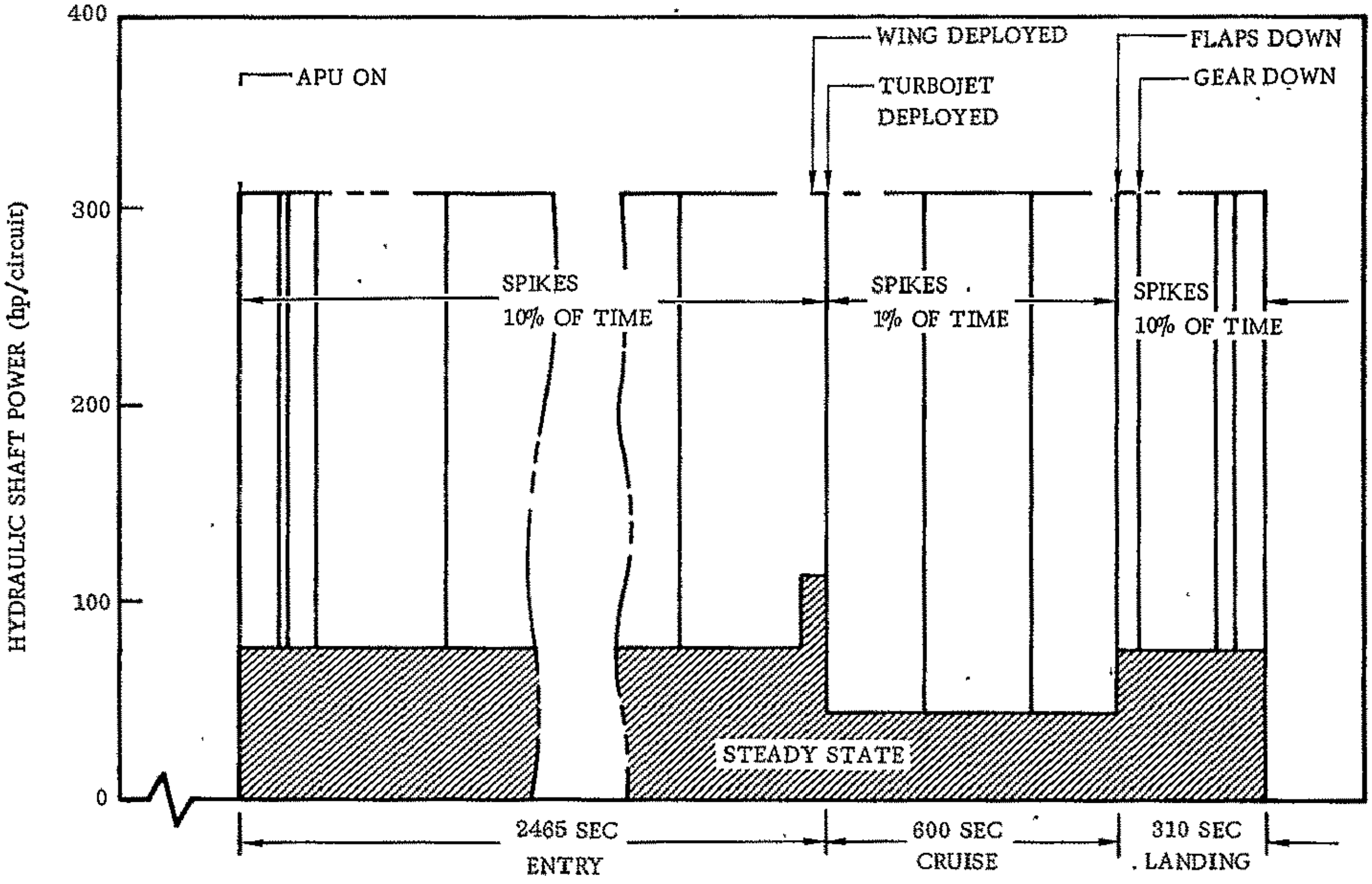


Figure 6-2. FR-4 Orbiter Hydraulic Power Profile

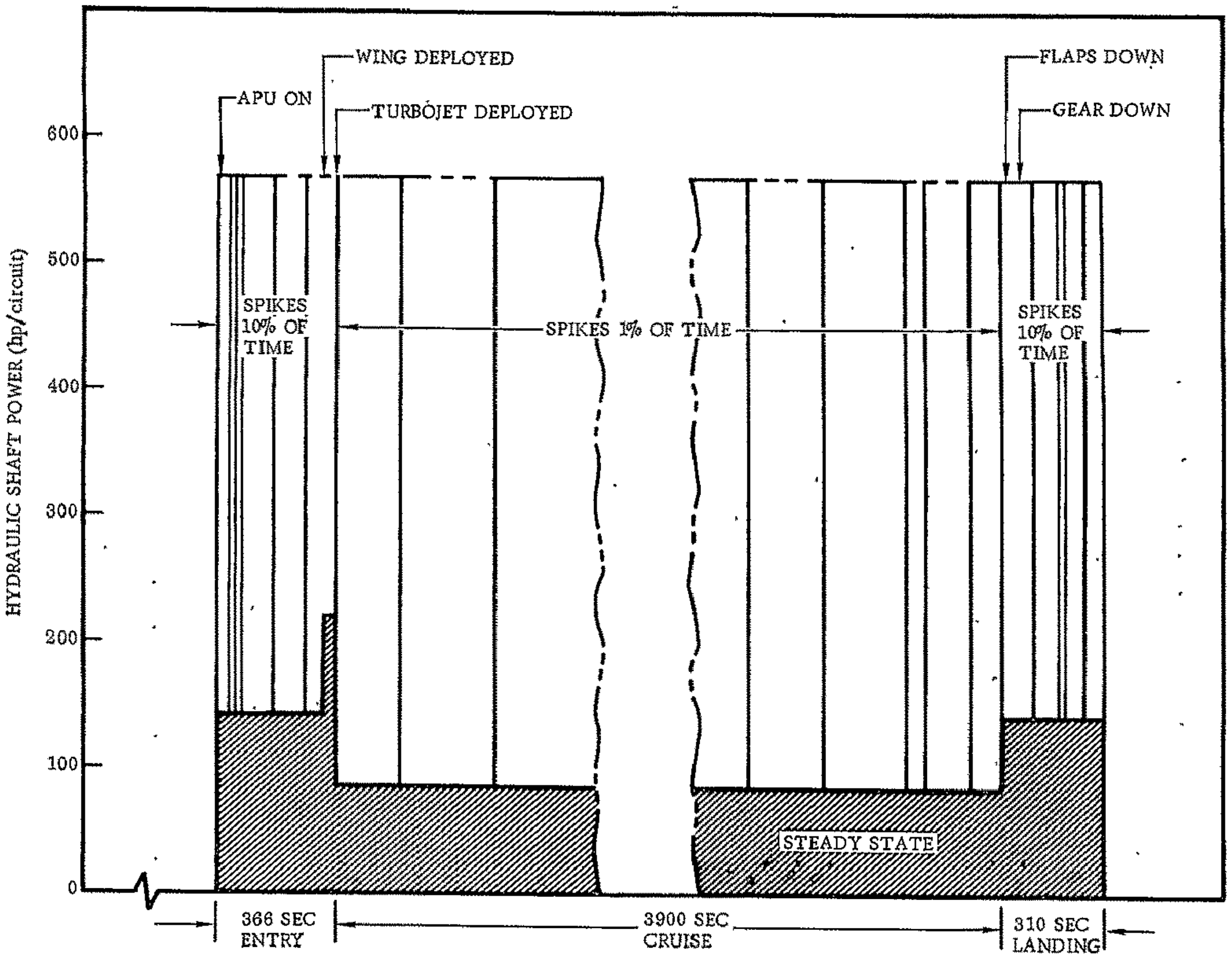


Figure 6-3. FR-3 Booster Hydraulic Power Profile

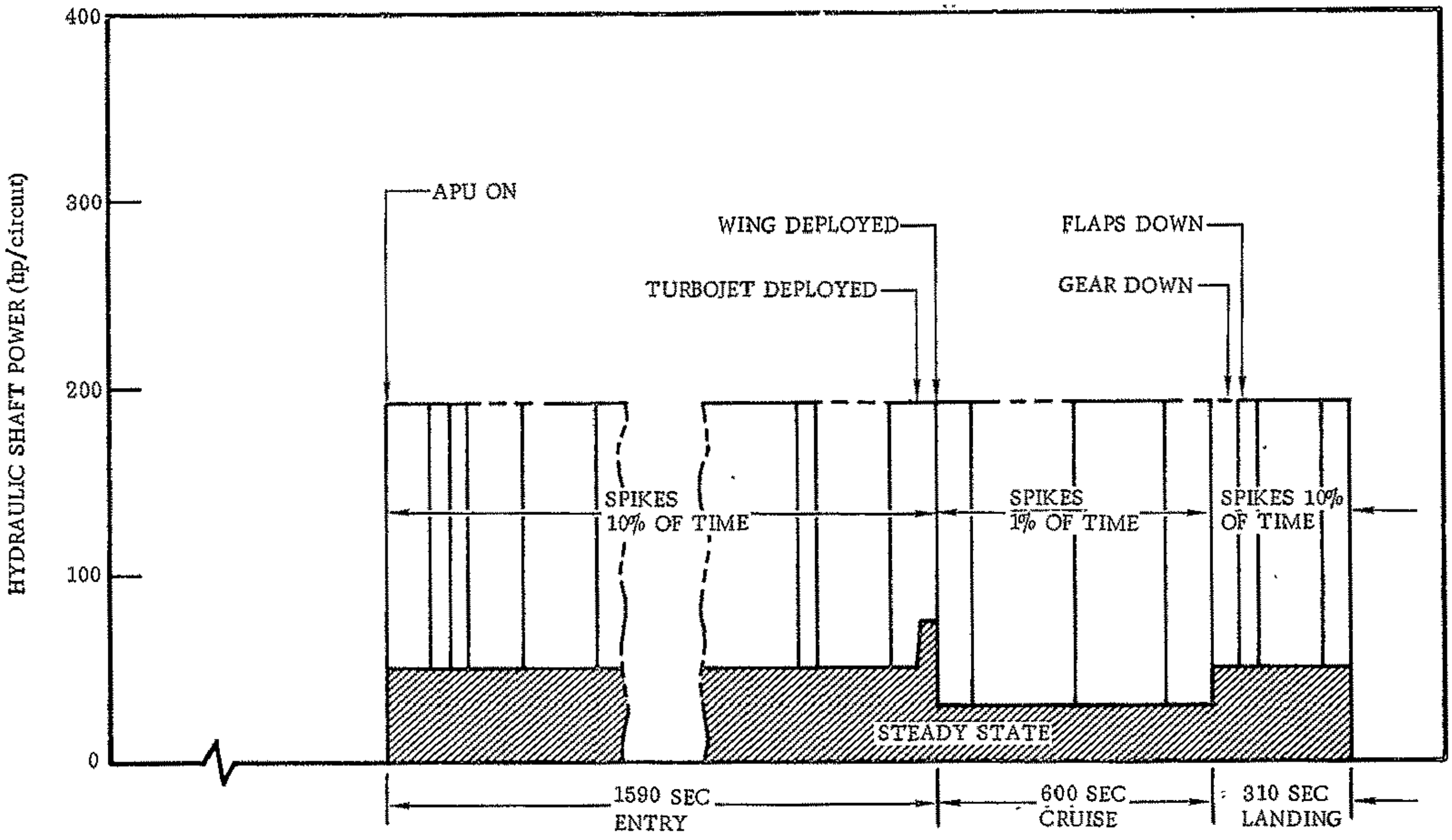


Figure 6-4. FR-3 Orbiter Hydraulic Power Profile

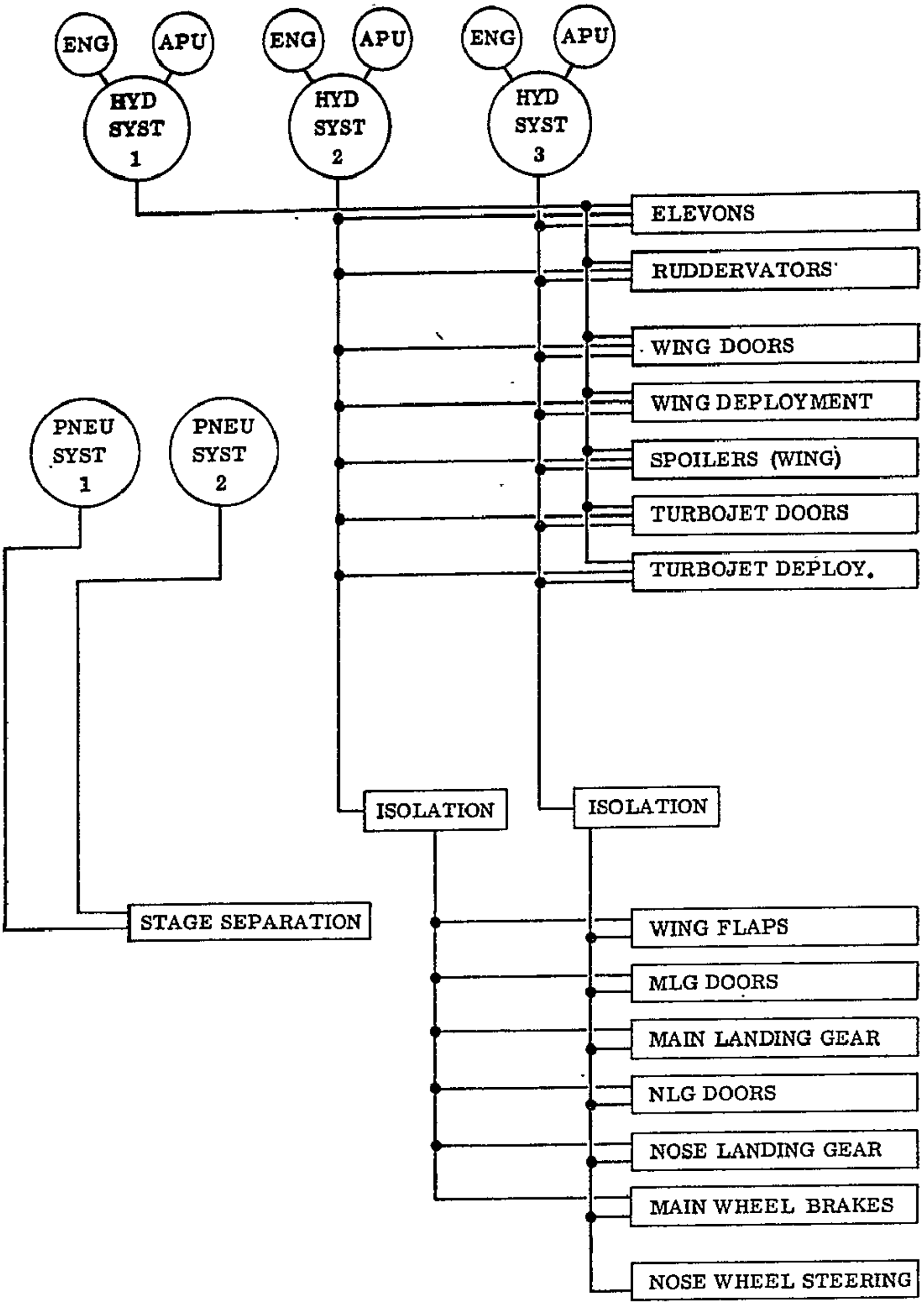

ENG
APU
HYD SYST 1
ENG
APU
HYD SYST 2
ENG
APU
HYD SYST 3
ELEVONS
RUDDERVATORS
WING DOORS
WING DEPLOYMENT
SPOILERS (WING)
TURBOJET DOORS
TURBOJET DEPLOY.
PNEU SYST 1
PNEU SYST 2
ISOLATION
ISOLATION
STAGE SEPARATION
WING FLAPS
MLG DOORS
MAIN LANDING GEAR
NLG DOORS
NOSE LANDING GEAR
MAIN WHEEL BRAKES
NOSE WHEEL STEERING

Figure 6-5. Actuation Subsystems, FR-4/FR-3 Booster

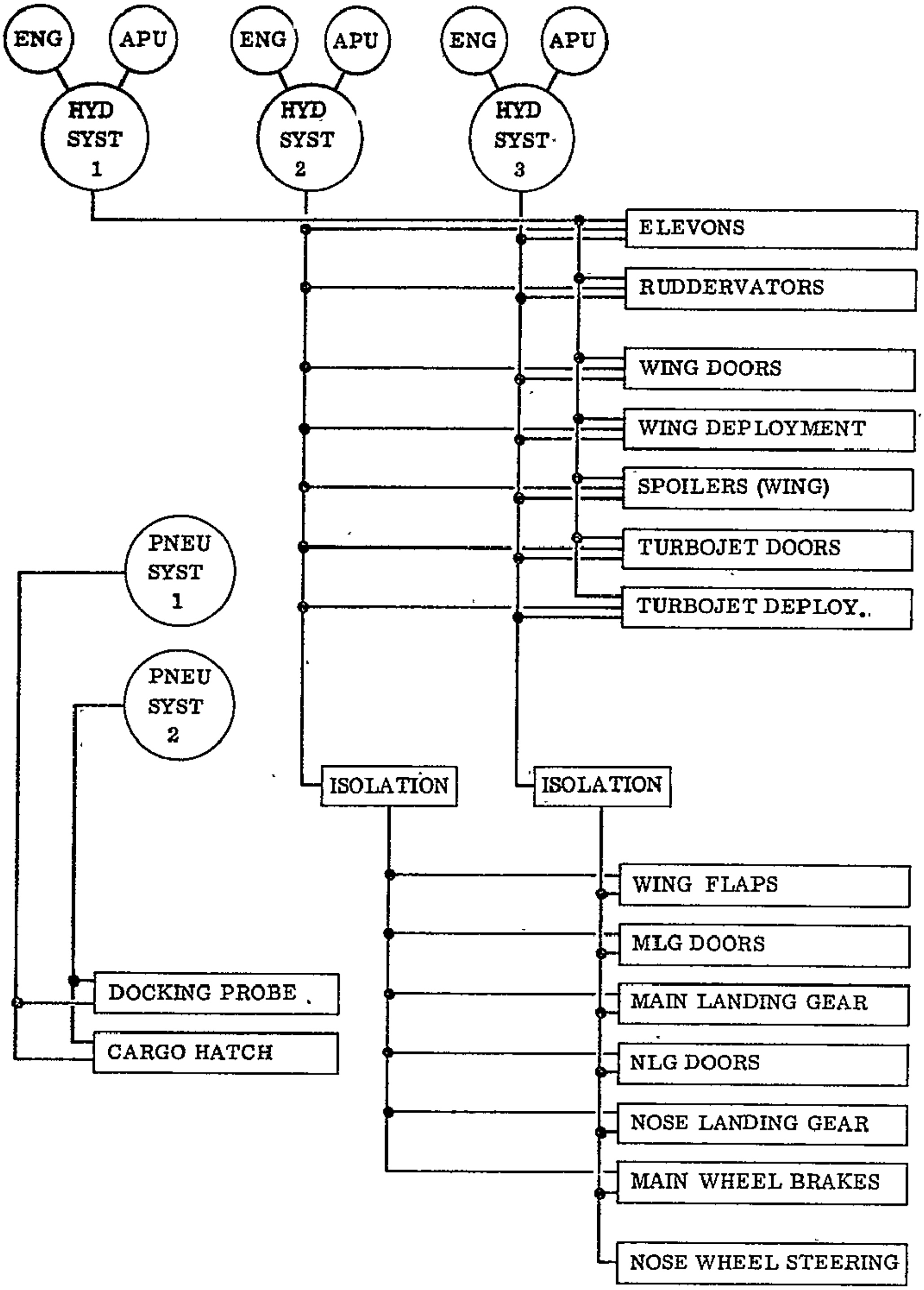

ENG
APU
HYD SYST 1
ENG
APU
HYD SYST 2
ENG
APU
HYD SYST 3
ELEVONS
RUDDERVATORS
WING DOORS
WING DEPLOYMENT
SPOILERS (WING)
TURBOJET DOORS
TURBOJET DEPLOY.
PNEU SYST 1
PNEU SYST 2
ISOLATION
ISOLATION
WING FLAPS
MLG DOORS
MAIN LANDING GEAR
NLG DOORS
NOSE LANDING GEAR
MAIN WHEEL BRAKES
NOSE WHEEL STEERING
DOCKING PROBE
CARGO HATCH

Figure 6-6. Actuation Subsystems, FR-4/FR-3 Orbiter

work functions from the essential work functions. In the event of loss of hydraulic circuits 2 and 3, emergency backup is provided; landing gear extension and limited operation of the flaps and wheel brakes are provided by independent accumulators.

Hydraulic system pressure is supplied by variable displacement hydraulic pumps. The hydraulic pumps are driven by three auxiliary power units (APU) during entry. Separate hydraulic pumps mounted on three turbojet engines are used as the power source during the cruise and landing phases. All hydraulic power generation equipment (including pumps, reservoirs, filters and accessories) is located forward of the nose landing gear in the subsystems compartment. Servicing of each hydraulic circuit can be accomplished at one central location, which is desirable from a maintenance standpoint. In general, the hydraulic system design will be based on current technology. No major development is anticipated for the principal components including pumps and servoactuators.

The type of hydraulic fluid to be used was not established. Preliminary analysis indicates that there is a tradeoff between a conventional Type II system (-65 to +275°F) using MIL-H-5606 hydraulic fluid and a Type III system (-40 to +450°F) using a high-temperature hydraulic fluid such as Oronite 70 or Oronite M2V silicate esters. A Type II system is desirable in the interest of minimizing development costs, but the cooling requirements may result in an unacceptable weight penalty. A Type III system offers potential weight savings along with somewhat increased development costs.

Certain miscellaneous work functions are required prior to activation of the hydraulic systems. These include stage separation, docking probe, and cargo hatch. Two independent pneumatic systems supply energy for activation of these work functions. Each pneumatic system will be supplied by high-pressure gas (air or nitrogen) contained in a storage bottle.

6.3 THERMAL CONSIDERATIONS

The heat generated during entry creates two distinct problem areas for the hydraulic system. These are concerned with thermal limitations of actuator rod-end bearings and decomposition of the hydraulic fluid.

Heat conducted through the structure to the actuator attach fittings poses a problem for the bearings in these fittings. To prevent these bearings from seizing, the maximum temperature must be limited to a reasonable level. Perhaps the most practical way of limiting bearing temperature is to conduct heat through the actuator body and piston rod into the hydraulic fluid. The hydraulic fluid could then be used to transport this heat to the heat exchanger. If the servovalve is designed with a somewhat larger-than-normal internal leakage, a continuous flow of hydraulic fluid would be available for carrying heat away from the servoactuator.

The hydraulic fluid leaving the servovalve is at return system pressure. This flow can be directed into a manifold that jackets the actuating cylinder. The inner surface area of this manifold must be adequate to permit the required heat transfer to the circulating fluid.

A relatively detailed layout of the actuator installation will be required before a meaningful heat transfer analysis can be applied and should be accomplished in the next study phase.

Using the hydraulic fluid to carry the heat away from these hot spots on the vehicle poses the second problem. As the temperature of the hydraulic fluid increases there is a potentially higher likelihood of degradation. Formation of sludges and varnishes can result in contamination of critical components. It therefore becomes necessary to limit the maximum temperature of the fluid. The residual hydrogen fuel remaining in the main engine tanks could be used to cool the hydraulic fluid. A more detailed study of the system installation is required to identify the seriousness and the resolution of these potential problems.

Further investigation is also required to define low-temperature fluid problems. Present concepts do not require operation of the hydraulic system in orbit. For long orbital times, hydraulic fluid in lines close to the hydrogen tank can become very viscous, and when the hydraulic system is pressurized it will restrict proper system operation. Probable solutions to this environmental condition will lie in careful line routing, line insulation, provision for a small, continuous fluid circulation, or heating of the lines prior to start.

6.4 ACTUATION SYSTEM FAILURE MODES ANALYSIS

A failure mode and effect analysis was conducted for two typical flight control actuation systems. Each system was broken down into the essential elements: power source, hydraulic pump; servo valve, linear actuator and plumbing.

The following component failures were considered:

a. Auxiliary Power Unit: failure to start, loss of fuel supply, fuel pressurization system malfunction.

b. Turbojet Engine: failure to deploy, engine inoperative, fuel supply malfunction.

c. Hydraulic Pump: loss of output flow or pressure.

d. Power Transfer Unit: sheared coupling shaft.

e. Servo Valve: loss of command signal, internal leakage.

f. Actuator: internal leakage.

g. Plumbing: external leakage resulting in loss of hydraulic fluid.

A failure mode and effects analysis requires the establishment of ground rules defining fail operational and fail safe conditions. These are:

a. Normal Operation. The actuation system is designed to produce 1.5 times nominal design hinge moment. The design deflection rate at no load is 30 deg/sec.

b. Fail Operational. The output torque and the deflection rate are equal to or greater than the nominal design values.

c. Fail Safe. The actuation system is capable of satisfying one of the following conditions:

   1. 50 percent of the nominal design hinge moment and 100 percent of the design deflection rate.
   2. 100 percent of the nominal design hinge moment and 50 percent of the design deflection rate.

The following failure modes and effects analysis is based on the preceding ground rules:

a. Scheme A - Two Tandem Actuators/Surface. Three independent power sources are provided to supply fluid power to the servo valves as shown in Figure 6-7. Two power transfer units are included to equalize the hydraulic load. Each power transfer unit consits of a motor-pump unit which supplies additional flow to hydraulic circuit 2 during peak loading conditions. Two linear tandem actuators convert hydraulic power to a linear output.

   The failure conditions and effects of failure are summarized in Table 6-1. It should be noted that a single failure of servovalve 2 or plumbing circuit 2 will result in a fail safe condition. In addition, certain double failure conditions of two servovalves or two plumbing circuits will result in an unsafe condition. This is not acceptable.

b. Scheme B - Three Actuators/Surface. Three independent power sources are provided to supply fluid power as shown in Figure 6-8. The actuation system includes triple redundant circuits consisting of a hydraulic pump, servovalve, and linear actuator.

   The failure conditions and effects of failure are summarized in Table 6-2. A single failure of any component or hydraulic circuit results in a fail-operational condition. In addition, any double failure condition results in a fail-safe condition. Scheme B was therefore selected as the baseline actuation system for operation of flight control surfaces.

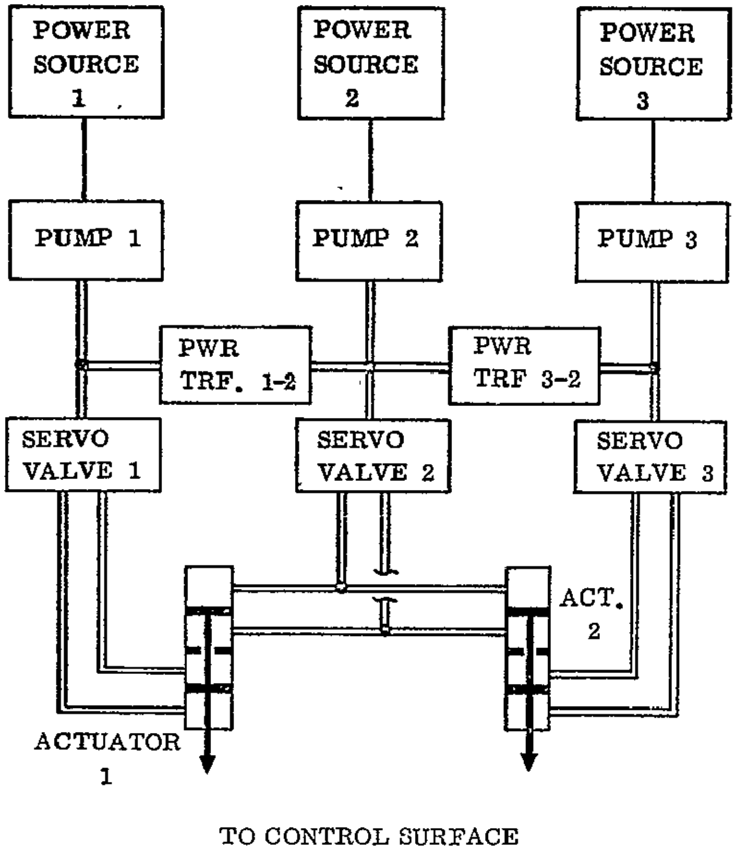

POWER SOURCE 1
POWER SOURCE 2
POWER SOURCE 3
PUMP 1
PUMP 2
PUMP 3
PWR TRF. 1-2
PWR TRF 3-2
SERVO VALVE 1
SERVO VALVE 2
SERVO VALVE 3
ACT. 2
ACTUATOR 1
TO CONTROL SURFACE

Figure 6-7. Typical Flight Control Actuation System (Scheme A)

Table 6-1. Flight Control Actuation System Failure Analysis (Scheme A)

Single Failure	Effect of Failure	
	Hinge Moment	Deflection Rate
1. Power Source (1, 2, or 3)	*	*
2. Hydraulic Pump (1, 2, or 3)	*	*
3. Power Trf. (1-2 or 3-2)	*	*
4. Servovalve (2)	†	*
5. Actuator (1 or 2)	*	*
6. Plumbing (2)	†	*
Double Failure		
1. Power Sources (1 and 2 or 2 and 3)	*	*
2. Hydraulic Pump (1 & 2 or 2 & 3)	*	*
3. Power TRF (1-2 and 3-2)	*	*
4. Servovalve (1 & 2 or 2 & 3)	‡	*
5. Actuator (1A and 2A)	†	*
6. Plumbing (1 & 2 or 2 & 3)	‡	*

* System Fail Operational; † System Fail Safe; ‡System Unsafe

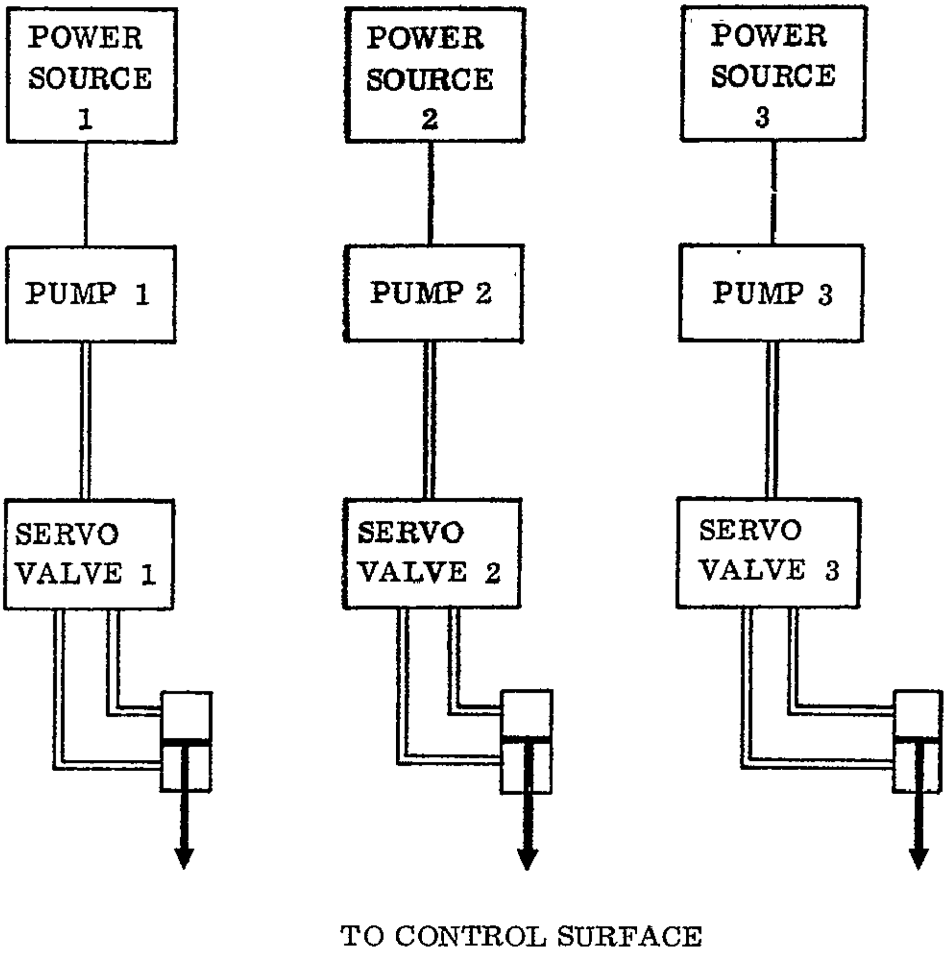

POWER SOURCE 1
POWER SOURCE 2
POWER SOURCE 3
PUMP 1
PUMP 2
PUMP 3
SERVO VALVE 1
SERVO VALVE 2
SERVO VALVE 3
TO CONTROL SURFACE

Figure 6-8. Typical Flight Control Actuation (Scheme B)

Table 6-2. Flight Control Actuation System Failure Analysis (Scheme B)

Single Failure	Effect of Failure	
	Hinge Moment	Deflection Rate
1. Power Source (1, 2, or 3)	*	*
2. Hydraulic Pump (1, 2, or 3)	*	*
3. Servovalve (1, 2, or 3)	*	*
4. Actuator (1, 2, or 3)	*	*
5. Plumbing (1, 2, or 3)	*	*
Double Failure		
1. Power Sources (1 & 2, 1 & 3 or 2 & 3)	†	*
2. Hydraulic Pump (1 & 2, 1 & 3 or 2 & 3)	†	*
3. Servoavlve (1 & 2, 1 & 3 or 2 & 3)	†	*
4. Actuator (1 & 2, 1 & 3 or 2 & 3)	†	*
5. Plumbing (1 & 2, 1 & 3 or 2 & 3)	†	*

---

* System Fail Operational; † System Fail Safe

6.5 SYSTEM WEIGHT

The hydraulic power generation and distribution system weight includes equipment required for generation and distribution of hydraulic power such as pumps, reservoirs, accumulators, filters, instrumentation, plumbing hydraulic fluid, and installation hardware. The total weight includes three hydraulic circuits for each vehicle element as summarized in Table 6-3.

Table 6-3. Hydraulic System Weight (Pounds)

ITEM	FR-4	FR-3	
		Booster	Orbiter
Pumps	457	820	280
Reservoirs and Heat Exchanger	322	580	190
Plumbing	286	510	170
Fluid	794	1420	480
Miscellaneous	151	270	160
Total	2010	3600	1280

SECTION 7

AUXILIARY POWER UNIT

7.1 GENERAL REQUIREMENTS

Power requirements for the auxiliary power unit (APU) were based on the aerodynamic control requirements and criteria of Section 4 and the selected baseline hydraulic system of Section 6. The unit must provide shaft horsepower on demand for operation of one hydraulic circuit during reentry. The work functions considered were: primary flight controls, wing deployment (if applicable), turbojet door and engine deployment, wing spoilers and flaps, main and nose landing gear and doors, main wheel brakes and nose wheel steering.

7.2 POWER AND ENERGY REQUIREMENTS

Power and energy requirements are as follows:

	2-Element System		3-Element System	
	Booster	Orbiter	Booster	Orbiter
Power-horsepower	567	190	318	318
Operating time/flight-seconds	366	1590	366	2465
Energy-horsepower-minutes	1260	1680	707	4300

7.3 TRADE STUDIES

Because of the great disparity in the power requirements for the basic vehicle electrical loads and the loads from Sections 4 and 6, it was decided to use separate power generation systems for the actuation of the aerodynamic control surfaces and miscellaneous functions. The establishment of 300-horsepower loads discouraged the study of electromechanical actuation methods. A preliminary investigation of various chemical-dynamic systems resulted in the selection of turbomachinery to provide high power output in a compact, lightweight package with relatively simple controls and minimum complexity.

Turbomachinery systems using both a hydrazine blend fuel and H_2-O_2 fuel were analyzed. As a baseline for the H_2-O_2 system, it was assumed that residual fuel from the main propulsion system would be used for the reaction control system and with appropriate integration could be made available to the APUs at a usable pressure for little or no added penalty. Both of the APU systems are turbine driven machines providing shaft power to two hydraulic pumps and use the products of combustion as the energy source. The systems are capable of providing 318 shaft horsepower split equally between two 6000 rpm output pads. Both systems were designed to be compatible with environmental temperatures of -65°F to 130°F.

7.3.1 HYDRAZINE BLEND FUELED APU

7.3.1.1 Gas Generator. Because of the -65°F temperature requirement, it is necessary to use a mixed hydrazine fuel (one containing freezing point depressants). One recommended fuel for this application is Sundstrand 70-20-10, a blend containing hydrazine, monomethyl hydrazine, and hydrazine nitrate. Since there are no known catalysts for carbon containing fuels, it is necessary to use a thermal type system where the retained heat in the chamber acts to initiate decomposition of the incoming fuel. The ignition method proposed herein utilizes solid propellant gas generators to heat the decomposition chamber prior to fuel injection, to accelerate the turbine, and to pressurize the fuel tank to provide inlet pressure to the fuel pump. This mode is presently being successfully employed in other Sundstrand turbine power systems.

7.3.1.2 System Description. The hydrazine system is composed of a gas generator, fuel tanks, turbine section, gear box, and accessory equipment. A functional schematic of the system is presented in Figure 7-1. The specific fuel consumption as a function of output shaft horsepower is presented in Figure 7-2. The mixture ratio is such that combustion chamber temperature is controlled to approximately 1500°F and chamber materials can be of conventional materials to ensure reliable long life.

The system is activated by an electrical signal which ignites a solid propellant start initiator and opens the fuel solenoid valve. The start initiator performs the following functions:

a. Accelerates the power unit to assist reaching full output in the required start time.

b. Heats the decomposition chamber to initiate fuel decomposition.

c. Begins fuel flow by pressurizing the fuel tank via the pressure tap on the diverging section of a turbine nozzle and by starting the fuel pump.

The hot gas from the fuel decomposition chamber is directed through converging-diverging supersonic nozzles to the turbine. The turbine drives two output shafts through a reduction gearbox. The pressure tap from the diverging section of a hot gas nozzle pressurizes the fuel tank to maintain fuel flow to the fuel pump inlet. Speed control is provided by fuel pump discharge pressure-actuated fuel control valves. The fuel pump is a centrifugal, Sundyne type with the characteristic of its discharge pressure varying as a square of its speed. The speed control pilot valve responds to variations in fuel pump pressure to cycle the fuel control valve between the on and off positions. Thus, the turbine sees a pulsed gas flow, and operation is at the design point for the fuel "on" condition. The fuel controls automatically match the hydraulic flow demand by providing a high percentage of "on time" with high hydraulic load and a low percentage of "on time" with reduced load. The control system varies the fuel

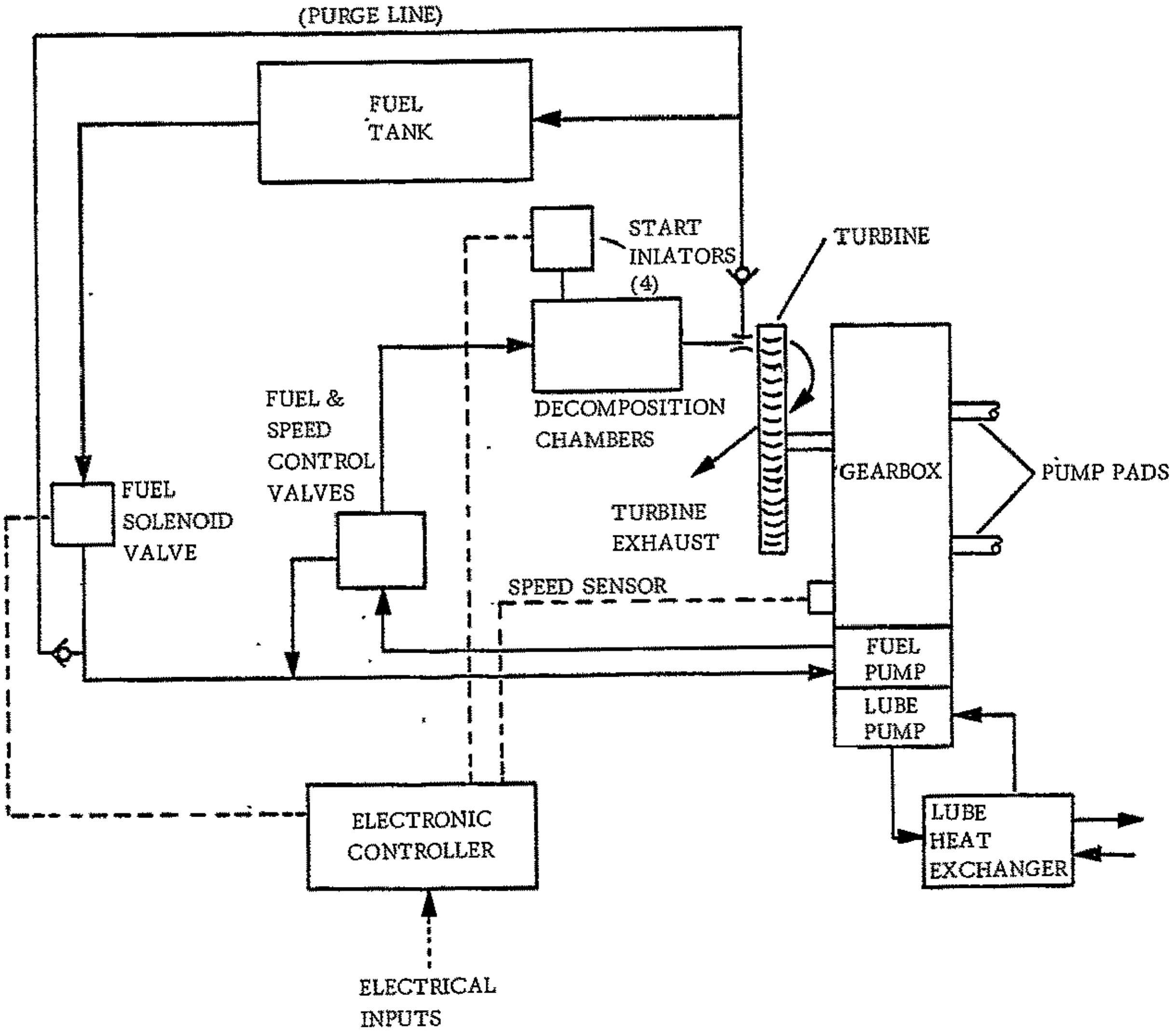

(PURGE LINE)
FUEL TANK
START INIATORS (4)
TURBINE
DECOMPOSITION CHAMBERS
FUEL & SPEED CONTROL VALVES
FUEL SOLENOID VALVE
GEARBOX
PUMP PADS
TURBINE EXHAUST
SPEED SENSOR
FUEL PUMP
LUBE PUMP
LUBE HEAT EXCHANGER
ELECTRONIC CONTROLLER
ELECTRICAL INPUTS

Figure 7-1. Hydrazine Blend Fueled FR-1-APU Schematic

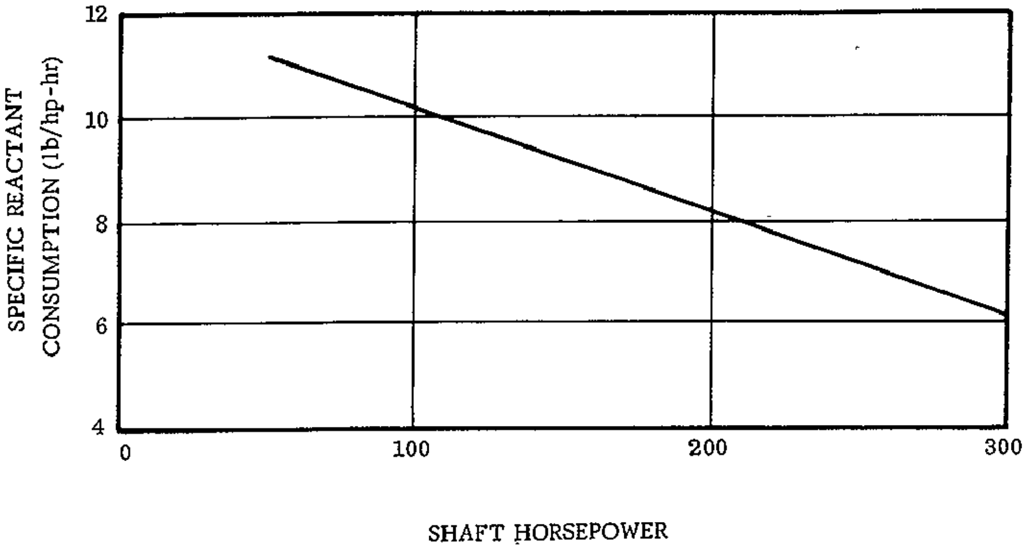


Figure 7-2. Hydrazine Consumption as a Function of Output Shaft Horsepower

consumption in proportion to the load demand, and operation is maintained within plus or minus 5% of rated speed, providing rapid response to load changes.

With complete expulsion of the fuel supply or intentional shutdown, the fuel passage areas are automatically purged of residual fuel prior to the soakback of heat from the hot sections of the power unit. The purge source is the nozzle tap warm gas which has been accumulated in the fuel tank.

The electronic controller operates in conjunction with the speed sensing magnetic pickup and the fuel solenoid valve to provide overspeed control of the unit. This control system allows continued power unit operation, at a safe speed, in case of failure of the normal speed control system. The controller processes the frequency signal from the magnetic pickup and opens and closes the fuel solenoid valve for speed control of the unit within a safe, elevated speed range.

Should failure of both the primary and secondary speed control occur, a mechanical shaft brake is provided to insure against destructive overspeed of the turbine.

Additional start grains are provided to permit three inflight restarts of the system.

The primary speed control consists of a pilot valve to monitor fuel pump outlet pressure and a bang-bang fuel control valve. Since the outlet pressure of a centrifugal pump varies as the square of its speed, the pump outlet pressure is a good speed indicator. When turbine speed exceeds a predetermined tolerance above the design speed, the pilot valve actuates, porting the pump discharge fuel pressure to the fuel control valve. The high pressure fuel closes the control valve and shuts off the fuel supply to the decomposition chamber. The turbine decelerates and as the fuel pump outlet pressure decreases, the valves return to their original positions and fuel flow is restored.

The secondary speed control system consists of a magnetic pickup which monitors turvine speed and a solenoid "bang-bang" fuel control valve. The magnetic pickup provides an automatic shut-down signal if an overspeed condition develops or can be used as a secondary "bang-bang" fuel control if the primary control should fail.

7.3.1.3 Technical Data. The technical data governing the design of this system, given in Table 7-1, define the per unit system characteristics for a two-element, two-stage vehicle system capable of a 50,000 lb in-orbit payload.

Table 7-1. Technical Data — Hydrazine Blend Fueled Unit

	2-Element System		3-Element System	
General System Data	Booster	Orbiter	Booster	Orbiter
Single System Weight, lb				
Turbine Power Unit	223	75	125	125
Fuel Tank	53	49	25	127
Fuel	202	245	95	634
Rated Shaft Power Output (total), hp	527	190	318	318
Hydraulic Pump Pad				
Quantity			2	
Rated Capacity	116	39	65 GPM Pump	
Speed			6000 RPM	
Operating Time, sec	366	1590	366	2465

	Common
Start Time (to full output)	5 sec
System Schematic	See Figure 7-1
Environmental Temperature	-65°F to +130°F
Altitude	
Normal Operation	200,000 ft to 15,000 ft
Limited Operation	Space, S.L.
Life	
Service Life	10 years
Operating Life	500 hours
Start Capability	450 starts
Restarts Per Mission	3
Maximum Surface Temperature	700°F

Table 7-1. Technical Data — Hvdrazine Blend Fueled Unit, Contd

Fuel	70/20/10 Hydrazine Blend
Constituents	70% Monomethyl Hydrazine 20% Neat Hydrazine 10% Hydrazine Nitrate
Instrumentation Provided	Lube Oil Pressure Turbine Exhaust Temperature
External Power Required (dc)	
Start Up	28 V 7.5 A
Operating	28 V 2.5 A
Lube Oil Heat Exchanger Requirements	
Heat to be Rejected	4000 Btu/min
Temperature to Heat Exchanger	200°F
Specific Propellant Consumption	Per Figure 5-28
@ 320 hp	5.9 lb/hp-hr
@ 80 hp	10.1 lb/hp-hr
Components	
Fuel Tank	
Type	Reusable (Configuration to be Defined)
Operating Pressure	250 psia
Decomposition Chamber	
Type	Thermal Regenerative
Operating Temperature	1950°R
Initiation Method	Solid Propellant Initiator
Operating Pressure	800 psia
Gearbox	
Ratio	9.5/1
Lubrication	Forced Lube
Pressurization	N_2
Turbine	
Type	Axial Flow Impulse, Two-Stage Reentry
Fuel Pump	
Type	Centrifugal (Sundyne)
Controls	
Primary	Fuel Pump Discharge Pressure Actuated Bang-Bang Valves
Over Speed Protection (Operating)	Electrical Solenoid
Over Speed Protection (Stop)	Shaft Brake

7.3.2 H_2-O_2 FUELED APU

7.3.2.1 Combustion Chamber. The combustion chamber selected for the H_2-O_2 version is based on previous Sundstrand experience with H_2-O_2 propellants. The combustor is a catalytic type cylindrical chamber culminating in an annular section mating with the full entry turbine. Combustion wall cooling is accomplished by use of excess hydrogen. Since starting time is not critical in this application, initiation of operation can be accomplished by a gradual increase of reactant flow to the design flow rate over a period of about 15 seconds. This startup time is a function of reactant inlet temperature.

7.3.2.2 System Description. This system is also comprised of a gas generator, turbine section, gearbox and accessory components. It differs, however, from the hydrazine system in that no fuel tanks are required, but additional controls are required to handle the additional fluid component and to condition the fluids from cryogenic temperature to a practical combustor inlet temperature. A typical functional schematic is shown in Figure 7-3.

A typical fuel consumption curve is presented in Figure 7-4. This curve is based on a fuel-rich mixture which will give a combustion gas temperature of 1500°F to allow use of conventional materials to provide long life and minimum maintenance. It is proposed to operate fuel-rich rather than oxidizer-rich to minimize the materials compatibility problem and to minimize performance loss resulting from operation in an off-stoichiometric condition.

The system is activated by an electrical signal which opens the hydrogen and oxygen shutoff valves supplying propellant to the system. Combustion is accomplished in a catalytic combustion chamber and the warm gas energy supplied to a two-stage turbine. The rate of propellant flow to the combustor is controlled during startup to provide adequate time to warm the chamber before full fuel flow is delivered. The rate of fuel flow to the combustor is controlled by two modulating servo valves. Both speed and combustor temperatures are monitored and controlled to maintain the proper turbine power level and operating conditions for all output loads. The turbine speed is maintained within ±5% of rated.

The turbine provides shaft power via a gearbox to the two 6000 rpm pump pads. The gearbox is force lubricated and lube oil flow is provided to an external heat exchanger for cooling.

A regenerator is provided in the turbine exhaust duct which, in conjunction with a bypass valve controls the hydrogen inlet temperature to approximately 500°R.

Should a failure occur in the speed control system, a mechanical shaft brake is provided to insure against destructive overspeed of the turbine. Multiple restart capability of the system is achieved by use of a catalytic type combustion chamber.

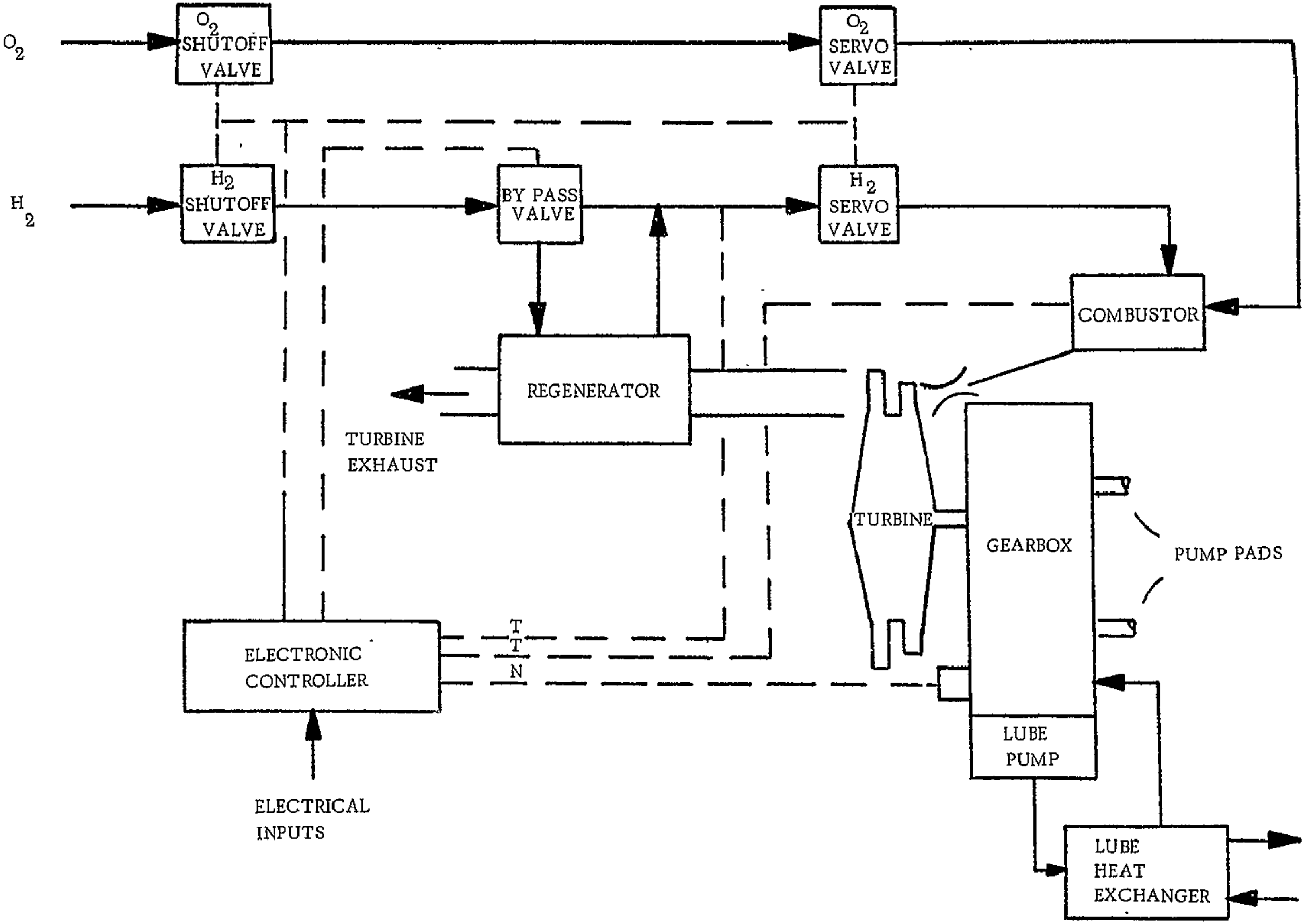

O_2
O_2 SHUTOFF VALVE
O_2 SERVO VALVE
H_2
H_2 SHUTOFF VALVE
BY PASS VALVE
H_2 SERVO VALVE
COMBUSTOR
REGENERATOR
TURBINE EXHAUST
TURBINE
GEARBOX
PUMP PADS
T
T
N
ELECTRONIC CONTROLLER
ELECTRICAL INPUTS
LUBE PUMP
LUBE HEAT EXCHANGER

Figure 7-3. H_2-O_2 FR-1 APU Schematic

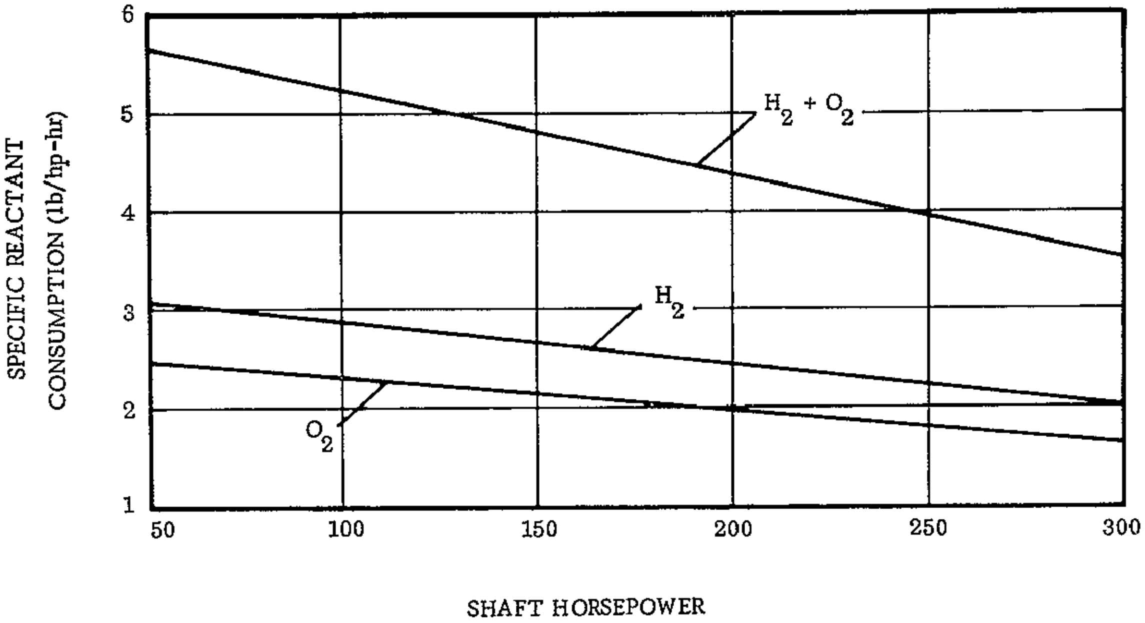


Figure 7-4. H_2-O_2 Consumption as a Function of Shaft Horsepower

Both speed and combustor temperature are controlled to maintain proper turbine operating conditions at all APU loads. Flow modulation of both propellants is achieved by individual modulating servo valves. In addition, shutoff valves are incorporated for positive propellant control during system startup and shutdown. A gaseous hydrogen bypass valve with a regenerator located in the turbine exhaust is employed to maintain the hydrogen temperature entering the combustor at 500°R to insure reliable chamber operation.

Hydrogen is received from the fuel system at the shutoff valve. The shutoff valve is pressure biased to prevent actuation when system pressure is inadequate. The H_2 shutoff valve module also incorporates the bypass system, which is positioned as a function of hydrogen inlet temperature to insure that this temperature is controlled to approximately 500°R.

Precise modulation of hydrogen flow for both speed (load) and temperature (mixture ratio) is accomplished by a jet-pipe torque motor stage actuating the floating second stage metering orifice.

The oxygen system is very similar to the hydrogen system with one exception: the oxygen shutoff valve provides for dumping oxygen from the O_2 lines at shutdown to prevent stoichiometric mixtures from being present at and immediately after shutdown.

A brief outline of the control logic and miscellaneous functions it performs includes:

a. Speed control
b. Turbine inlet temperature control
c. Combustor inlet temperature control
d. Regenerator bypass temperature control
e. Malfunction protection
    1. Overspeed
    2. Overtemperature
    3. Undertemperature
f. Programmed start ignition for low temperature conditions
g. Logic, start, stop, automatic disabling, and ground operation disabling.

If the hydrogen supply temperature was at least 400° R, the system would be simplified by eliminating the H_2 temperature control loop consisting of the hydrogen bypass valve, remote control logic, and regenerator. This is a very desirable approach as the system weight will be reduced by approximately 35 lb; most of this is the regenerator weight.

7.3.2.3 Technical Data. The technical data governing the design of this system, given in Table 7-2, define the per unit system characteristics for a two-element, two-stage vehicle system capable of a 50,000 lb in-orbit payload.

Table 7-2. Technical Data — Hydrogen and Oxygen Fueled Unit

	2-Element System		3-Element System	
General System Data	Booster	Orbiter	Booster	Orbiter
System Weight, lb				
Turbine Power Unit	267	94	150	150
Regenerator	53	19	30	30
Hydrogen	51.7	78	29	191
Oxygen	42.0	63	23.6	155
Hydraulic Pump Pad				
Quantity	116	41	2	
Rated			65 GPM Pump	
Speed			6000 RPM	
Envelope Required				
Turbine Power Unit			17 in. dia. × 18 in. long	
Regenerator			0.3 ft^3	
Rated Shaft Power Output (total), hp	567	190	318	318
Operating Time, sec	366	1590	366	2465

Table 7-2. Technical Data — Hydrogen and Oxygen Fueled Unit, Contd

	Common
Start Time (To Full Output)	30 sec
System Schematic	See Figure 7-3
Environmental Temperature	-65°F to +130°F
Altitude	
Normal Operation	200,000 ft to 15,000 ft
Limited Operation	Space, S.L.
Life	
Service Life	10 years
Operating Life	500 hours
Start Capability	450 starts
Restarts per Mission	3
Maximum Surface Temperature	700°F
Instrumentation Provided	Lube Oil Pressure Turbine Exhaust Temperature
External Power Required (dc)	
Start Up	28V 7.5A
Operating	28V 2.5A
Lube Oil Heat Exchanger Requirements	
Heat to be Rejected	1100 Btu/min
Temperature to Heat Exchanger	200°F
Specific Propellant Consumption	Per Figure 5-28
@ 320 hp	3.4 lb/hp-hr
@ 80 hp	5.4 lb/hp-hr
Fuel Supply Conditions to Unit	
H_2 Supply	100 psig 170°R
O_2 Supply	100 psig 500°R
Components	
Combustion Chamber	
Type	Catalytic
Operating Temperature	1500°F
Initiation Method	Palladium Catalyst
Operating Pressure	85 psia
Gearbox	
Ratio	9.5/1
Lubrication	Forced Lube

Table 7-2. Technical Data — Hydrogen and Oxygen Fueled Unit, Contd

Turbine	Two Stage, Axial Flow Impulse
Type	Two Stage, Axial Flow Impulse
Controls	
Primary	H_2 and O_2 Servo Valve Shutoff Valves (Speed and Temperature Control)
Overspeed Protection (Stop)	Shaft Brake
Exhaust Duct Size	4.5 in. Diameter

7.4 SYSTEM SELECTION

Because of the integration possibility and the potential weight savings, the H_2-O_2 APU system was chosen as the baseline system to provide power to the functions described in Sections 4 and 6. To satisfy the redundancy requirements, three independent systems are provided, each rated at 318 shaft horsepower. Considerable development work has been accomplished on H_2-O_2 systems and has culminated in a prototype system designed for the now defunct DynaSoar system. Further development will be necessary to satisfy and demonstrate the reliability requirements of the space shuttle systems. The cryogenic conditioning required will complicate the system unless it is accomplished in the normal development of the reaction control supply system from which the H_2-O_2 fluids are to be obtained.

7.5 TOTAL SYSTEM WEIGHT ESTIMATES

The total system weight includes the necessary redundancies to meet the failure criteria. A summary is given in Table 7-3.

Table 7-3. Summary of Total System Weight Estimates

Item	2-Element System		3-Element System	
	Booster*	Orbiter	Booster	Orbiter
Power Unit	320	113	360	180
Fuel	94	141	105	346
Installation	130	46	146	73
Total/Unit	544	300	611	599
Total/Vehicle**				
System Total	1632	900	1833	1797
	2532		3630	

---

* Weights are for two boosters.

**Includes three systems per vehicle.

SECTION 8

THERMOSTRUCTURAL DESIGN

8.1 INTRODUCTION

During the sizing procedure that was used to develop the FR-3 and FR-4 configurations shown in Figures 3-4, 4-4, and 4-16, (Volume II) one interim configuration was selected for use in a point design of the thermostructural subsystem. This configuration, which is an element in the equal element FR-1 vehicle, is identified as the T-18. The T-18 configuration is shown in Figure 8-1, and its thermostructural design is reported in Section 8.4. It presents a conceptual approach to the airframe design that is generally valid for the FR-3 and FR-4.

Development of the general structural arrangement is paced by the vehicle configuration in the fuselage bays containing propellant storage where the fuselage cross-section consists of the outer heat shield, a space for frames and stiffeners, and the propellant tank wall. Since both the heat shield and tank wall can be designed to provide load paths for overall body shear, either can be made the primary airframe structure depending on the location of the frame and stiffener attachment. If the heat shield is designed as a "hot" airframe structure, thermal expansion slip joints in the heat shield will have to be eliminated to provide transfer of body shear and longitudinal bending continuity. Then the airframe will be subjected to the uncertainties associated with extensive variation of the thermal environment surrounding the outer surface during reentry. If the propellant tanks are designed as the airframe structure, pressure vessel discontinuities and cryogenic contractions provide complication. With either arrangement, the heat shield and the propellant tanks must be integrated to form the vehicle structure. Because stiffened cryogenic pressure vessels are more state-of-the-art than 20,000 sq ft stiffened radiative heat shields, the stiffened propellant tank approach is used throughout the baseline structural arrangements reported in Sections 8.4 and 8.5. Use of the heat shield as a "hot" primary airframe structure is a definite candidate concept, especially in the booster structure, and a conceptual arrangement of an FR-3 booster hot structure is shown in Paragraph 8.5.

Using stiffened propellant tanks as an integral part of the structural arrangement, the point design was developed for a vehicle 204 feet long with a body volume of 131,000 cu.ft. As shown in Table 8-1, this body size is somewhat larger than the FR-4 vehicle (6.8 and 18.1 percent) and the FR-3 orbiter (32.3 percent) but close enough to provide a representative design. The FR-3 booster, however, is significantly larger in size than the point design. Because of this and other configuration differences, a different structural arrangement is presented for the FR-3 booster in Section 8.5.

Table 8-1. Configuration Geometry Comparison

	T-18 Point Design	FR-3		FR-4	
		Orbiter	Booster	Orbiter	Booster
Length	204 ft	-12.3%	+ 2.8%	- 6.3%	- 2.4%
Width	35 ft	-11.5%	+17.4%	- 6.0%	- 1.8%
Height	30 ft	-13.2%	+23.0%	- 6.2%	- 2.0%
Volume	131,313 ft^3	-32.3%	+79.0%	-18.1%	- 6.8%
Wetted Area	19,308 ft^2	-23.4%	+34.8%	-12.5%	- 4.6%
LO T/W	1.39	Same	Same	+ 5.0%	+ 5.0%

Significant configuration differences other than geometry occur in areas of propellant storage, propulsion, and the thermal protective system.

The propellant storage system for the orbiter consists of structurally integral tanks forward and aft of the payload compartment with several small auxiliary tanks installed along the payload compartment bottom. In the FR-4 booster, the arrangement consists of one long integral tank divided by a common bulkhead into oxygen and hydrogen storage. Design problems encountered in both of these arrangements are discussed with structural concepts presented in the T-18 design (Sections 8.4.5 through 8.4.7) which consists of two integral tanks and one large divided auxiliary tank. The weight analysis in Section 11 is based on weight estimates of the specific vehicles.

Variation in the propulsion arrangement occurs in the number and thrust level of the engines. The orbiters have three engines at 400,000 lb thrust each. The FR-4 booster has nine engines at 400,000 lb thrust each. The T-18 point design is for a vehicle with two engines of 1,000,000-lb thrust each. Although there is considerable dispersion in the engine arrangement and total thrust load, the point design presents an approach to structurally supporting the engines and distributing the thrust load that is valid for all of the vehicles. The weight estimates for the various vehicles account for differences in thrust level and engine arrangement.

The T-18 point design presented in Section 8.4 is significantly modified in one section to reflect a design approach consistent with less demanding requirements. Because the crossrange requirement for the FR-3/-4 vehicle is less than for the FR-1, reentry temperatures are reduced. This permits a reduction in the use of exotic materials in the heat shield, reduced insulation weight and simplification of the thermal protective system support structure. An FR-3/FR-4 design is reported in Section 8.4.14, Thermal Protective System.

Propellant Tank Requirements (Sections 8.4.5 through 8.4.7)

a. Maximum ullage pressures shall be:

LO_2	23.0 psig limit
LH_2	28.5 psig limit

b. Hydrostatic pressures shall be based on:

LO_2	Tanks separated by sequence valves
LH_2	All tanks ducted together

c. Insulation shall consist of:

LO_2	None
LH_2	Internal

d. Main tanks are integral with the airframe structure.

e. Provision shall be included for entering all tanks for maintenance.

Centerbody Structure (Section 8.4.8)

a. Payload is 15 ft in diameter by 60 ft long, and weighs 50,000 lb.

b. Payload bay doors must open a 15 by 60-ft area with a 1 ft of clearance all around.

c. Payload doors must be remotely operable at 1 g or zero g.

d. Wing pivot is based on F-111 design.

e. Wing pivot critical design loads are:

Vertical shear = 418,000 lb ultimate per side (landing condition)

Bending moment = 165×10^6 in-lb ultimate (subsonic gust condition)

Aft Body (Section 8.4.9)

a. Provisions shall be made for installation of two rocket engines and the distribution of thrust to the vehicle airframe.

b. Engine output is:

Number installed — 2

Thrust per engine — 1,200,000 lb limit

Gimbal angle — ±3 deg (corner = 5 deg)

Actuator moment — 670,000 in.-lb limit

c. Provisions shall be made for vehicle holddown and lateral stability with the vehicle in the vertical position.

d. Critical holddown support loads are:

Upper fitting, P_x = 1,255,000 lb compression ultimate

Lower fittings, P_x = 696,000 lb compression ultimate

Side load, P_y/P_z = 17,500 lb ultimate

e. Support for the V-tail stabilizers shall be provided.

f. Critical stabilizer loads at the stabilizer root per side area:

Moment = 35,600 in.-lb ultimate

Shear = 364,000 lb ultimate

Torsion = 21,800,000 in.-lb ultimate

g. Aft closure heat-shield protection for the thrust section shall be provided:

Pressure differential = 2.7 psig limit

Maximum allowable deflection = 2.0 in.

Maximum allowable temperature in thrust section compartment = 500°F

h. Interstage connections shall be supported.

Interstage Connections (Section 8.4.13)

a. Interstage connections will provide attachment of the orbiter stage to the booster stage.

b. Loads:

Maximum thrust transfer into orbiter is 2,960,000 lb ultimate at booster burnout, with 230,000 lb ultimate transverse loads from the resulting couple.

Maximum stage separation linkage loads are 775,000 lb ultimate in the X direction, and 116,000 lb ultimate in the Z direction.

c. Load Factors:

The ultimate load factor is 1.25 × limit load for thrust-associated loads.

The ultimate load factor is 1.40 × limit load for separation linkage loads. A dynamic amplification factor of 1.50 is added to separation linkage loads.

d. Interstage connections are designed for minimum penalties to the orbiter vehicle. Wherever possible, the penalties were placed on the booster.

e. Interstage connections (including stage separating linkage) are completely reusable except for replacement of pyrotechnic release devices.

Wing (Section 8.4.11)

a. Dimensional Requirements:

Area = 1760 ft^2 per vehicle

Aspect ratio = 9.25

Airfoil — Root = NACA 4421
Tip = NACA 4418

M.A.C. = 13.9 ft

Span-exposed = 63.75 in.

Deployment = 80 deg

b. Environment requirements:

Subsonic speeds

Stowed temperature = 200°F maximum

c. High-lift devices:

Trailing edge "inverted" type full span, single-slotted flap.

25 percent of wing area.

Screw-type actuators

d. Spoilers are hinged type, 10 percent of chord, 25 percent of outer span.

e. Leading edge is fixed, 10 percent of wing chord.

f. Trailing edge is 30 percent of wing chord.

Stabilizer, V-Tail (Section 8.4.12)

a. Dimensional requirements:

Area = 1800 ft^2 per vehicle (exposed)

Aspect ratio = 1.375

Airfoil - Root = NACA 4412 Mod.

Tip = NACA 4410 Mod.

Leading edge radius = 6.00 in. constant

Span = 35.25 ft

b. Environment requirements:

Orbiter — Nose temperature = 2750° F

Remaining maximum temperature = 1650° F

Booster — maximum temperature = 1250° F

c. Rudder travel is 10 deg down, 45 deg up.

Thermal Protection System (Section 8.4.14, FR-3 and FR-4 Configurations)

a. The thermal protection system (TPS) includes the provisions made to accommodate aerodynamic heating during launch and reentry, heating from the engines, and the cryogenic temperatures of LO_2 and LH_2.

b. The TPS for the FR-3 and FR-4 vehicles shall be designed to meet the requirements defined in the remainder of this section (these requirements are similar to those for the FR-1 vehicle discussed in Section 8.1, except for generally lower heat shield maximum temperatures and some changes permitted by the lower temperatures). The orbiters for the FR-3 and FR-4 vehicles have the same TPS requirements. The boosters for the two vehicles are of different configurations and therefore have somewhat different TPS requirements.

c. The TPS shall be designed to withstand the thermal, aerodynamic, acoustic, and vibration environments during 50 cycles of launch and reentry with a minimum of refurbishment required.

d. The design maximum heat shield temperatures for the orbiters are shown in Figures 4-5, 4-6, 4-7, and 4-8 of Volume IV.

e. The design maximum heat shield temperatures for the FR-3 booster are shown in Figure 4-55 of Volume II.

f. The design maximum heat shield temperatures for the FR-4 booster are similar to those shown in Section 3.6.3 for a two-stage vehicle.

g. The temperature of the TPS support structure and the vehicle structure shall not generally exceed 200°F at any time during the life of the vehicle.

h. A pressure barrier on the cold side of the insulation on the base of the orbiter vehicles shall seal the base TPS from the interior and sides of the vehicles. The total leakage area past the barrier shall not exceed 5 in^2 over the entire base of each vehicle. The pressure barrier shall be designed to carry a 2.5 psi ultimate planform loading on the base of the vehicle without exceeding the 0.2 percent yield stress of the barrier material.

i. The vertical deflection of the outboard ends of the TPS support beams under the wings shall not exceed 1.0 in. relative to the center of the beams when the base of the vehicle is subjected to a 1.0 g planform loading.

j. Except for the noses and leading edges, the base heat shields on the orbiters shall be designed so that the Δp across the heat shield shall not exceed 0.1 psi when the heat shield temperatures are within 400° of their peak values. The maximum rate of change of the local shock pressure on the base is 0.02 psi per second sustained for 10 seconds when the heat shields are within 400°F of their peak temperatures.

k. The orbiter base heat shields shall be designed to withstand a continuous Δp of 0.1 psig during all cycles without exceeding 0.2 percent creep in the heat shield materials.

l. When 400°F below their maximum design temperatures, the base heat shields shall be able to withstand 1.5 psi ultimate Δp without exceeding the 0.2 percent yield stress of the heat shield material.

m. No pressure barrier other than the heat shields is required on the top and sides of the orbiters and boosters. The leakage area shall not exceed 0.10 in^2 for any 100 ft^2 of these heat shields. The Δp for these heat shields is 1.0 psi ultimate when within 200°F of maximum temperatures and 1.5 psi ultimate at other times. While loaded to these ultimate Δp's, the heat shield materials shall not exceed the 0.2 percent yield stress.

n. No pressure barrier other than the heat shields is required on the base of the booster. The leakage area shall not exceed 0.10 in^2 for any 100 ft^2 of these heat shields. The Δp for these heat shields is 1.5 psi ultimate when within 400°F of their maximum temperatures and 2.5 psi ultimate at other times. While loaded to these ultimate Δp's, the heat shield materials shall not exceed the 0.2 percent yield stress.

o. The posts and TPS support structure shall withstand loads applied to the outer ends of the posts equivalent to a ±40 g amplified vibratory load from the mass of the heat shields in a direction parallel to the axis of the vehicle. Equivalent loads in other directions shall be 20 g.

p. The design requirements for the insulation of the cryogenic fluids are discussed along with the design approach in Sections 8.4.5, 8.4.6, and 8.4.7.

q. Design allowables for the principal heat shield materials are shown in Table 8-2.

8.2.2 STRUCTURAL DESIGN CRITERIA. This section includes the major structural design criteria used with the FR-1 FR-3, and FR-4 configurations. The basis for this section is contained in Aerospace Corporation document Structural Design Criteria for the Space Transportation (STS) Study, dated 25 June 1969, revised 9 July 1969. Deviations from this document were made where values could be either calculated or obtained from specifications such as the MIL-A-8860 series.

Table 8-2. Properties of Heat Shield Materials

Material	Room Temperature Properties				0.2% Creep Stress (ksi) for 50 Cycles (10 hr)	Properties at Maximum Temperature During Cycle				
	UTS (ksi)	0.2% Yield Strength (ksi)	Elongation in 2 in.	Maximum Temp During Orbiter Mission (°F)		UTS (ksi)	0.2% Yield Strength (ksi)	Elongation in 2 in.	ρ (lb/in.^{3})	Principal References
Hastelloy X	100	45	35%	1500	10	46	24	30%	0.295	MIL-Hdbk 5; Aerospace Structural Metals Hdbk
Coated T-222 Tantalum	103	88	15% (in 1 in.)	3063	~2.5	21	20	40% (in 1 in.)	0.604	Interim Tech. Rpt. 6, AF33(615)-3935 contract, McDonnell Astronautics Co.
Coated C-129Y Columbium	69	58	11%	2400	~4.0	28	25	2%	0.343	GDC-ERR-1272; Martin ER 13117 (Contract AF33(657)-11539); GDC-ERR-1345
TD NiCr	125	82	10%	2200	5.0	~11	8.0	1%	0.306	AFFDL-TR-68-130 Part I, McDonnell Douglas
HS-188 (Haynes)	140	60	50%	2000	1.2**	14.3	13.7	30%	0.333	GDC-ERR-1272; GDC-ERR-1345; Technical Paper*
6Al-4V Titanium	134	126	8%	800	60	90	78	8%	0.160	MIL-Hdbk 5; Aerospace Structural Metals Hdbk

* A New Oxidation Resistant, High Strength, Ductile, Cobalt Base Alloy, R. B. Herschenroeder of Union Carbide (Haynes), presented 17 October 1967 for Metallurgical Society of AIME.

** Approximately same for L-605.

Design limit load factors for prelaunch and ground-air-ground cycle conditions are shown in Table 8-3. Limit proof, yield, and ultimate factors of safety are given in Table 8-4. Limit pressure differentials for cabin, tankage, venting, and several structural items are shown in Table 8-5. Dynamic amplification factors applied to limit loads obtained by rigid body analyses are:

CONDITION	FACTOR
MAXIMUM αq	1.20
MAXIMUM βq	1.20
LANDING, MAIN GEAR	1.20
LANDING, NOSE GEAR	1.60
SEPARATION	1.50

Material strengths were based on room temperature properties, except where elevated temperatures are encountered (i.e., no advantage was taken of increased material strength at temperatures below room temperature).

Material allowable stresses for fatigue were determined for a fatigue life equal to four times the number of cycles expected during the total life of the spacecraft. Maximum allowable stresses for creep were determined using creep rates equal to twice the nominal creep rates under predicted conditions of load, time, and temperature. The tanks were not dependent on pressurization for their structural integrity during any handling operations or during fueling.

All spacecraft will be free of flutter, buzz, or divergence at dynamic pressures up to 1.32 times the maximum dynamic pressure expected to be encountered at any flight Mach number. All external surfaces of the spacecraft will be free of destructive flutter at dynamic pressures up to 1.5 times the maximum local dynamic pressure expected to be encountered at any Mach number within the envelope of operating conditions.

The structure will endure a service life of 100 missions (except the thermal protection system, for which the life is 50 missions). Temperature histories for various spacecraft locations are presented in Section 4 of Volume IV.

As a guide in the selection of materials, Table 8-6 indicates typical minimum gage thicknesses established for aluminum, titanium, superalloys, and refractory metals encompassing various structural configurations.

8.2.3 LOAD ANALYSIS. Loads for major FR-1, FR-3, and FR-4 structural components were calculated including body, wing, fin, and landing gear loads for various ground and flight conditions. The net loads were determined by computer programs that handle airload and mass distributions, cruise and booster thrust vectors, concentrated loads, and translational and rotational inertias. The vehicle is in quasi-static

Table 8-3. Design Limit Load Factors

Phase	Condition	N_x	N_y	N_z	Remarks
Prelaunch	Air Transporation	+0.33 -1.00	± 0.75	-2.50 +1.00	Dry Weight
	Land Transporation	± 1.00	± 0.50	-3.00	
	Water Transporation	± 0.50	± 0.60	-2.50	
	Jacking	± 0.50	± 0.50	± 2.00	
	Hoisting	+ 2.00	± 0.70	± 2.00	
	Towing	± 0.50	± 0.25	± 1.50	
	Erection	-1.50	± 0.25	± 0.50	
	Ground Wind	-1.00	0.00	0.00	Unpressurized full tanks
Boost	Liftoff	+1.40	0.00	0.00	
	Max αq	+1.85 -2.03	± 0.10	± 0.40	Orbiter Booster
	Max βq	+1.85	± 0.45	± 0.10	
	Booster Burnout	+4.00	0.0	0.0	
	Separation -	1.10	0.0	0.0	
	Separation +	+1.65	0.0	0.0	Orbiter
		-1.10	0.0	-0.5	Booster
Orbit	Plane Change	+0.50	0.00	0.00	Orbiter only
	De-Orbit	+0.50	0.00	0.00	
Entry	Hypersonic Pullout	-0.10	± 0.10	-4.00	Booster
		-0.40	± 0.10	-2.00	Orbiter
	Supersonic Maneuver	-0.10	± 0.10	-2.00	
	Wing Deployment	-0.15	± 0.10	-1.25	
	Subsonic Gust	-0.10	± 0.50	-1±1.32	Per MIL-A-8861
Landing	2-Point Landing	-0.50	± 0.25	-2.00	12 fps sink speed
	3-Point Landing	-0.50	± 0.25	-2.00	
	Taxiing	±1.60	± 0.25	-1±0.75	
Ferrying	Taxiing	±1.60	± 0.25	-1±0.75	
	Takeoff Run	-0.30	± 0.10	-1±0.75	

Note: A plus (+) sign indicates acceleration forward, starboard, and down.

Table 8-4. Factors of Safety

	Factor of Safety				
Item	Limit	Proof	Yield	Ult.	Applied On
Main Structure	1.00	-	1.10	1.40	Air and transverse inertial loads
Main Structure	1.00	-	1.10	1.25	Thrust and axial inertial load
Thrust Beams	1.00	-	1.20	1.50	Thrust loads
Propellant Tanks	1.00	1.10	1.15	1.50	Ullage + Hydrostatic Head
Personnel Compartment	1.00	1.33	1.50	2.00	Pressure
Pneumatic Vessels	1.00	1.33	1.50	2.00	Pressure
Hydraulic Vessels (With Ullage)	1.00	2.00	3.00	4.00	Pressure
Hydraulic Vessels (No Ullage)	1.00	1.33	1.50	2.00	Pressure
Hydraulic & Pneumatic Lines and Fittings	1.00	2.00	3.00	4.00	Pressure
Propellant Supply & Vent	1.00	1.33	1.50	2.00	Pressure
Joints and Fittings	1.15	-	1.30	1.65	Loads
Landing Gear	-	-	1.00	-	Loads
Fatigue	4.00	-	-	-	Number of cycles
Creep	2.00	-	-	-	Creep rates
Flutter and Divergence	1.32	-	-	-	Dynamic pressure (NASA-SP-8003)
Panel Flutter	1.50	-	-	-	Dynamic pressure (NASA-SP-8004)
Thermal Loads	1.10	-	-	-	Thermal gradients
Miscellaneous Non-flight Loads	1.00	-	1.10	1.25	Loads

Table 8-5. Pressure Differentials (Limit)

Item	ΔP (psig)	Remarks
Cabin Bulkheads	+10.0	
Cabin Walls	+10.0	
Tankage		
LH_2	28.5	Maximum ullage pressure
LO_2	23.0	Maximum ullage pressure
Aft End Closure Bulkhead	2.7	
Heat Shields	-1.07	Bottom
Heat Shields	-0.71	Sides and top
Venting	+1.0 -0.5	All cavities except cabin

Table 8-6. Minimum Gage Selection Guide for Various Materials

Material	Flat Skin	Single Corrugation	Skin Corrugation		Sandwich		Integrally Stiffened Skin
			Skin	Corrug.	Facings	Core	
2219-T81	0.020	0.016	0.016	0.016	0.010	0.002	0.016
2024-T86	0.020	0.016	0.016	0.016	0.010	0.002	0.016
Ti-6Al 4V	0.016	0.016	0.016	0.012	0.010	0.002	0.016
Ti-8Al-1Mo-1V	0.016	0.016	0.016	0.012	0.010	0.002	0.016
718 Ni Alloy	0.010	0.010	0.010	0.010	0.010	0.002	0.016
Hastelloy X	0.010	0.010	0.008	0.008	0.010	0.002	0.016
L605	0.010	0.010	0.008	0.008	0.008	0.002	0.016
TD Nickel Chromium	0.010	0.010	0.008	0.008	0.008	0.002	0.016
Cb752	0.020	0.012	0.016	0.012	0.010	0.002	0.016
C129Y	0.020	0.012	0.016	0.012	0.010	0.002	0.016
T-222	0.016	0.012	0.012	0.012	0.010	0.002	0.012

equilibrium in all cases. Rigid body analysis was used. Details relative to airloads, mass distributions, and net loads are given in Volume IV, Section 5.

8.3 MATERIALS

The range of temperature to be encountered by the space shuttle vehicle is very wide — varying from the temperature of liquid hydrogen to that of the nose-cap stagnation point during reentry, approximately 3500° F. As a consequence of this range of temperatures, many types of materials are required for a vehicle to withstand the various flight conditions. The internal tankage with its necessary insulation is designed for cryogenic temperature, while the primary load-carrying structure is designed for temperatures near 200° F. The external heat shields will experience temperatures up to 3200° F, and the nose section stagnation region will be exposed to an equilibrium temperature of 3500° F.

The selection of materials involves such criteria as specific strength, specific modulus, creep strength, resistance to fatigue, fracture toughness, oxidation resistance, compatibility, corrosion resistance, thermal expansion, thermal conductivity, specific heat, emittance, fabricability, weldability, and cost. Specific material selection is described in the following paragraphs and summarized in Table 8-7. Typical mechanical properties for candidate materials are shown in Table 8-8.

The primary structure of the crew compartment and forward equipment bay from Station 0 to Station 21 will be fabricated from aluminum alloy 7075-T6. Shear loadings as well as loadings on the bulkheads, longerons, and stringers are sufficiently low that the choice of aluminum can be made on the basis of providing the lowest unit weight while supplying good corrosion and fatigue resistance and low notch sensitivity.

The body section housing the fly-back engine bays extending from Station 21 to Station 38.3 will be fabricated from Ti-8Al-1Mo-1V. The titanium alloy was selected because the temperatures at the structure surrounding the fly-back engine door openings are higher than the normally accepted 250°F for aluminum. A less-than-perfect seal around the door opening and TPS could cause hot gas seepage into the compartment. The alloy affords greater stiffness at low structural weight (E/ρ) when compared with aluminum alloys.

The bulkheads at Stations 21 and 38.3 will not be subjected to elevated temperatures and therefore afford the choice of aluminum alloy 7079-T6 forgings.

The transition structure between the Station 38.3 bulkhead and the LO_2 tank will be fabricated from aluminum alloy 7075-T6. This section is not subject to elevated temperatures and the choice offers an optimum strength-to-weight ratio.

The materials considered for the LO_2 tank were aluminum alloy 2219 and Inconel 718. Tradeoff studies indicate that 2219 is preferable on the basis of the least airframe stability weight. Aluminum 2219 is also favored for the auxiliary propellant tanks. Similarly, for the LH_2 tank, 2219 is preferred on the basis of resisting body bending loads when compared to Alloy 718.

Table 8-7. Material Selection Summary

	Candidate Materials	Leading Candidates	Max. Design Temperature	Back-Up Abort Material	Area of Application
Aluminum	2014 2024 2219 7075	2219 7075	200°F		Bulkheads, Longerons, Webs, LO_2 Tank
Composites	Alum-Boron Titanium-Boron Graphite-Epoxy Boron-Epoxy	Al-B Boron-Epoxy	600°F	Inconel 718	Frames, Longerons, Stringers, Bulkheads
Titanium	Ti-8Al-1Mo-1V Ti-6Al-4V Ti-5Al-2.5Sn	Ti-8-1-1 Ti-6-4	800°F	Inconel 718	Fly-Back Engine Bay, Aft Body Structure Thrust Structure, Heat Shields
Superalloys	Rene 41 Inconel 625 Inconel 718 L-605 HS-188 Hastelloy X	Inconel 718 L-605 HS-188	1200°F (For Inconel 718) 1900°F (For L-605) 2000°F (For HS-188)	L-605 (For Inconel 718) TD NiCr (For L-605 and HS-188)	Heat Shields, Structural Fittings,
DSM	TD Ni TD NiCr	TD NiCr	2200°F	C-129Y	Heat Shields
Columbium	Cb-752 C-129Y	C-129Y	2500°F	T-222	Heat Shields
Tantalum	Ta-10W T-111 T-222	T-222	3100°F	ZrB_2 + SiC C-C	Heat Shields Leading Edges
	HfB_2 + SiC ZrB_2 + SiC	ZrB_2 + SiC	3500°F	C-C	Leading Edges, Nose Cap

Table 8-8. Mechanical Properties of Candidate Materials

Alloy	Design Temp.	Mechanical Properties									
		Room Temperature					Design Temperature				
		$F_{TU} \times 10^3$ psi	$F_{TY} \times 10^3$ psi	$E \times 10^6$ psi	e in 2 in. (%)	$\alpha \times 10^{-6}$ in/in/°F	$F_{TU} \times 10^3$ psi	$F_{TY} \times 10^3$ psi	$E \times 10^6$ psi	e in 2 in., (%)	$\alpha \times 10^{-6}$ in/in/°F
2219-T81 ρ = 0.102 lb/in^3	200°F	60	45	10.5	6	12.4	55	42	10.3	7	12.7
7075-T6 ρ = 0.101 lb/in^3	200°F	76	65	10.3	11	12.7	66	61	9.8	12	13.2
Ti-8Al-1Mo-1V ρ = 0.158 lb/in^3	600°F	145	135	17.5	10	4.6	135	90.5	15.1	13	5.1
Ti-6Al-4V ρ = 0.160 lb/in^3	600°F	134	126	16.0	8	4.5	103	83.8	13.1	15	5.8
Alum-Boron (50 v/o unidirectional, long) ρ = 0.100 lb/in^3	500°F	165	165	33.7	-	5	155	155	30.1	-	6.1
L-605 (Haynes 25) ρ = 0.330 lb/in^3	1900°F	145	77	29.8	18	5.8	14	13	20.8	23	9.6
TD Ni Cr ρ = 0.306 lb/in^3	2200°F	126	86	21.4	15	6.8	11	8	8.7	1	9.9
C-129Y VH-101 coating ρ = 0.343 lb/in^3	2500°F	69	58	15.5	11	3.7	28	25	9.5	3	4.6
T-222 R-512C coating ρ = 0.604 lb/in^3	3100°F	103	88	26	15	3.5	21	20	-	40	4.4
ZrB_2 + SiC (ManLabs Material VIII) ρ = 0.158 lb/in^3	3800°F	30	30	76	1	4.2	30	30	63	1	4.2

Titanium alloy 8-1-1 was selected as the material for the aft body structure, which extends from Station 176.2 to 189.5. The selection was based on providing a 15-percent weight saving over aluminum alloy construction, the need for less insulation since the design temperature is 600°F as opposed to 200°F for aluminum, and greater stiffness at low structural weight.

The thrust structure consists of four-foot-deep beams that support the engines at Station 189.5. These beams will be fabricated from Ti-8-1-1 with additional cap material of aluminum-boron unidirectional composite. This combination provides additional beam rigidity with little structural weight increase.

The launch holddown structure and the stabilizer support structure consisting of beams, struts, and fitting will be fabricated from Ti-8-1-1. The choice was based on providing low structural weights.

The forward stage interconnect attach point consists of aluminum alloy 2219 for the shear webs and longerons. These structures are integral with the LO_2 tank skirt and hence it would be most desirable to maintain material compatibility. The frames will be aluminum-boron caps on aluminum webs. High frame loads make the use of Al-B advantageous while reducing cap sections and providing increased stiffness.

The rear stage interconnect attach points are integral with the thrust structure. Therefore, using Ti-8-1-1 would maintain material compatibility. For the linkage and interconnect fitting, Alloy 718 will be used to take advantage of a high E/ρ. The material possesses high temperature strength and excellent oxidation resistance up to 1200°F.

The radiative TPS includes the heat shields, support structure, and insulation. The insulation materials will be in the form of fibrous blankets. Where the temperatures do not exceed 1600°F, Johns-Manville Microquartz will be used. For temperatures between 1600 and 2700°F, Johns-Manville Dyna-Flex is proposed. Union Carbide's ZYF-100 zirconia felt will be used for temperatures up to 3200°F. For temperatures above 3200°F, adequate fibrous materials are not commercially available, but thoria felt appears to be an excellent candidate.

The support structure at the cold side of the insulation will be primarily Ti-6Al-4V or Ti-8Al-1Mo-1V with some unidirectional graphite-epoxy composite. The material for the hot side of the support is generally the same as the heat shields. The support post material generally transitions to Ti-6-4 at the cold end of the posts.

The heat shield materials are reviewed in Table 8-2. For temperatures below 800°F, Ti-6Al-4V is the primary candidate. This alloy was selected over Ti-8-1-1 in areas of thin gage, exposed surfaces to avoid the uncertainty of saline stress corrosion cracking. As an alternative material, if higher heating prediction methods prove more accurate, Alloy 718 would provide an adequate safety margin for temperatures up to 1200°F.

The selection of alloys for use up to 2000°F perhaps offers the most latitude. L-605 (Haynes 25) is generally regarded as the most acceptable material when compared with Inconel 625 and Hastelloy X. Tests indicate that Inconel 625 exhibits better tensile properties, but this is overshadowed by lower oxidation resistance and a lower residual fatigue limit (Reference 8-1).

Hastelloy X has a significantly higher resistance to oxidiation, but the mechanical properties (including creep rate) are less advantageous than L-605 (Reference 8-2). A relatively new alloy, Haynes HS-188, is being examined under a Convair-funded IRAD program (Reference 8-3). When compared with L-605 and Hastelloy X, this material offers greater forming latitude due to its higher elongation, while maintaining about the same tensile strength and creep resistance as L-605. In addition, HS-188 has approximately 4.6 times the oxidation resistance of L-605 and over 1.2 times that of Hastelloy X at 2000°F. This is due primarily to addition of lanthanum, which modifies the protective oxide scale and results in a very tenacious oxide that is impervious to oxygen diffusion.

TD NiCr is the most favorable material for use up to 2200°F for the lightly loaded TPS structure. Recent tests at Convair show that new heats of material exhibit elongations of 5 percent at 2200°F. Therefore, the embrittlement difficulty formerly encountered is reduced to acceptable levels. The reduced transverse properties present no problem for the loading conditions of the vehicle. Convair has shown a minimum abort condition failure of 3 cycles at 2200°F and 6000 psi for the FDL-5A vehicle trajectory.

Coated columbium alloys are the most preferred choice for the 2200 to 2500°F regime. C-129Y is preferred over Cb-752 because of its easier formability, better weldability, and slightly better mechanical properties (Reference 8-3). The oxidation-resistant coatings are silicides. Sylvania R-512E and Vac-Hyd VH-101 have shown the best results. A minimum of 35 cycles can be expected using the present trajectories.

For temperatures between 2500 and 3200°F, tantalum alloy T-222 with silicide coatings R-512C or VH-105 have been found most acceptable. The maximum cyclic life has not yet been determined, but a minimum of 10 cycles would be acceptable.

For areas of the vehicle that will experience temperatures above 3200°F, ceramic materials such as zirconium diboride ($ZrB_2 + SiC$) are considered leading candidates. These materials are still in the pilot application stage and must have proven scale-up capability. Arc plasma tests have been made at heat fluxes well above those to be experienced by the present space shuttle vehicle configurations.

8.4 SELECTED THERMOSTRUCTURAL CONCEPTS

8.4.1 GENERAL DESCRIPTION. The approach to the structural design of the shuttle vehicles, both booster and orbiter, has been one of conventional structural arrangements and materials and state-of-the-art fabrication methods. However, high-strength materials such as composites, titanium, and other heat-resistant alloys have been used where thermal and/or loading conditions showed them to be superior to other materials. The major structural assemblies shown in Figure 8-2 are:

a. Forward fuselage, including the crew compartment and turbofan engine bays.

b. LO_2 and LH_2 tanks.

c. Center section, including payload bay (orbiter only), wing pivot bulkhead, and landing gear bulkheads.

d. Thrust structure.

e. Tank transition structures.

f. Stabilizers and wings.

g. TPS and support structure.

Most structures, including the LO_2 and LH_2 tanks, have been designed for a thermal environment of 200°F or less. This temperature is maintained at the structural envelope during reentry by the TPS. The exceptions are the aerodynamic stabilizer, wing, and orbiter payload bay doors. During reentry, the swing wing is stowed in its compartment and is not subjected to elevated temperatures. The stabilizers and payload doors are designed from materials (such as Inconel and titanium alloy, respectively) that can withstand elevated temperatures and maintain their structural integrity at reduced but acceptable levels.

The fuselage section forward of the LO_2 tank is a semi-monocoque shell that contains the crew compartment, equipment bay, and turbofan engine compartments. Whereas the crew compartment is designed to be pressurized, the remainder of the forward fuselage is vented to ambient conditions. A major bulkhead at Station 38.3 supports the turbofan engine pivots and forms the structural joint for the transition to the LO_2 tank. The design of this fuselage section adheres to the classical methods of shell stiffening through the use of frames, bulkheads, and stiffeners. Longerons have been arranged to carry and redistribute concentrated loads that occur in the vicinity of the cockpit windshield and entrance and near the turbofan engine doors.

The LO_2 and LH_2 tanks are both designed to form an integral part of the load-carrying vehicle structure. They are fusion-welded assemblies of wide circular rings that form the tank skin and frames. At each end, ellipsoidal domes form the closure bulkheads. The tanks are joined to the other structural sections of the vehicle by means of transition skirts.

The overall structural arrangement shows an excellent adaptability to a logical manufacturing breakdown into major assemblies at the transition structures.

8.4.2 CREW COMPARTMENT. The crew compartment is located at the forward end of the vehicle between Stations 0.0 and 21.0 (see Figure 8-2). The compartment contains the crew station and equipment bays for installation of the vehicle subsystems. Since an external heat shield covers the entire outer surface of the compartment, the structural operating temperature is maintained below 200°F, permitting the use of aluminum as the structural material. Load intensity is compatible with the efficient use of aluminum.

The vehicle structure in this area is arranged to satisfy the following requirements:

a. Distribute into the airframe the effect of lower surface airload of 220 psf at an ultimate safety factor of 1.4.

b. Maintain structural integrity under airframe bending loads of about 100 lb/in.

c. Provide pressurizable section for the crew station.

d. Provide load redistribution structure forward of the engine compartment access doors. These doors are located aft of Station 21.0.

From the vehicle nose to the aft end of the crew station Station 14.0 in Figure 8-2), the least-weight structural arrangement that meets the preceding requirements consists of a frame and longeron stiffened skin. The frame/skin combination carries the internal cabin pressure, supports the heat shield, and distributes the heat shield load into airframe body shear. The longerons react bending loads and stiffen the skin at major changes in shell contour. They also provide longitudinal continuity through the cabin area where it must be cut out for a transparent canopy and for crew ingress. These same longerons run continuously into the engine cutout area aft of Station 21.0.

Aft of the crew cabin area (Station 14.0 to 21.0) the primary loads are body shear and bending, so the semi-monocoque type of structure used in the cabin area is transitioned into a skin/stringer arrangement. Frames support the heat shield and distribute the heat shield airloads into vehicle body shear. The stiffened skin reacts the shear and the stiffeners and longerons react the bending loads.

8.4.3 TURBOFAN ENGINE COMPARTMENT. When not deployed, the turbofan engines are stowed in bays contained in that section of the fuselage extending from Station 21.0 to Station 38.3. The engine pylon pivot fittings are mounted on the bulkhead at Station 38.3. Each of the openings in the fuselage, through which the engines pass when in transit, is closed by a set of symmetrically arranged door assemblies. (See Figure 8-3.)

The shell of this fuselage section is of the semi-monocoque type. Its skins, stiffeners, frames, and longerons are fabricated from Ti-8Al-1Mo-1V alloy. This alloy was selected because of high temperatures at the structural skin in the vicinity of the engine doors. Since the insulation of the TPS is necessarily interrupted in this area, these temperatures will be substantially higher than 200°F (the normal level during reentry).

The cross-section of the engine bay fuselage section is bell-shaped. Longerons have been arranged at the lower outboard corners, along the bottom on the centerline, at the junction of the upper round portion and the flat sides, and along the sides of the upper engine door opening. These longerons are either continuous beyond Station 21 and Station 38.3 or terminate after having transferred their loads into the fuselage structure either forward or aft of the engine compartment.

Stretch-formed frames are attached directly to the inside of the skin and spaced approximately 12 inches apart. Where these frames are interrupted by the engine door openings, they are fastened to a frame that surrounds each door opening. At each door hinge point (four per door), the fuselage frames are spaced five inches apart and the door hinge and actuator loads are introduced into the shell. The door hinge fittings are mounted external of the skin and are backed up by supports nested between the closely spaced fuselage frames.

The shell of the fuselage consists of contoured thin-gage skins with zee-shaped stiffeners attached to the outside. Stiffener spacing is approximately eight inches. This arrangement permits stiffener continuity without fuselage frame cutouts, resulting in a more favorable fixity for the stiffener/effective-skin column when loaded in compression.

Stiffeners and skins continue forward of Station 21.0 into the forward fuselage section. (See Section 8.4.2.) They terminate at the bulkhead at Station 38.3, and structural continuity is provided by the transition structure to the LO_2 tank. (See Section 8.4.4.)

The fuselage section is divided into three separate bays (one for each of the turbofan engines) by a transverse, beam-stiffened web. This web, of a shallow vee-section, is joined at its apex by a vertical web that extends downward on the centerline of the fuselage and is attached to the lower center longeron. The outboard edges of the transverse web are fastened to the intermediate longerons.

To react and distribute the loads induced in the shell by the large door openings, a major frame (or bulkhead) is located at Station 21.0. Another bulkhead at Station 38.3 serves a similar purpose. Additionally, this bulkhead supports the engine pylon pivot fittings and provides the attach flange for the LO_2 tank transition structure.

The bulkheads are assembled from frame sections machined from 7079-T6 aluminum alloy forgings and from a stiffened sheet-metal web.

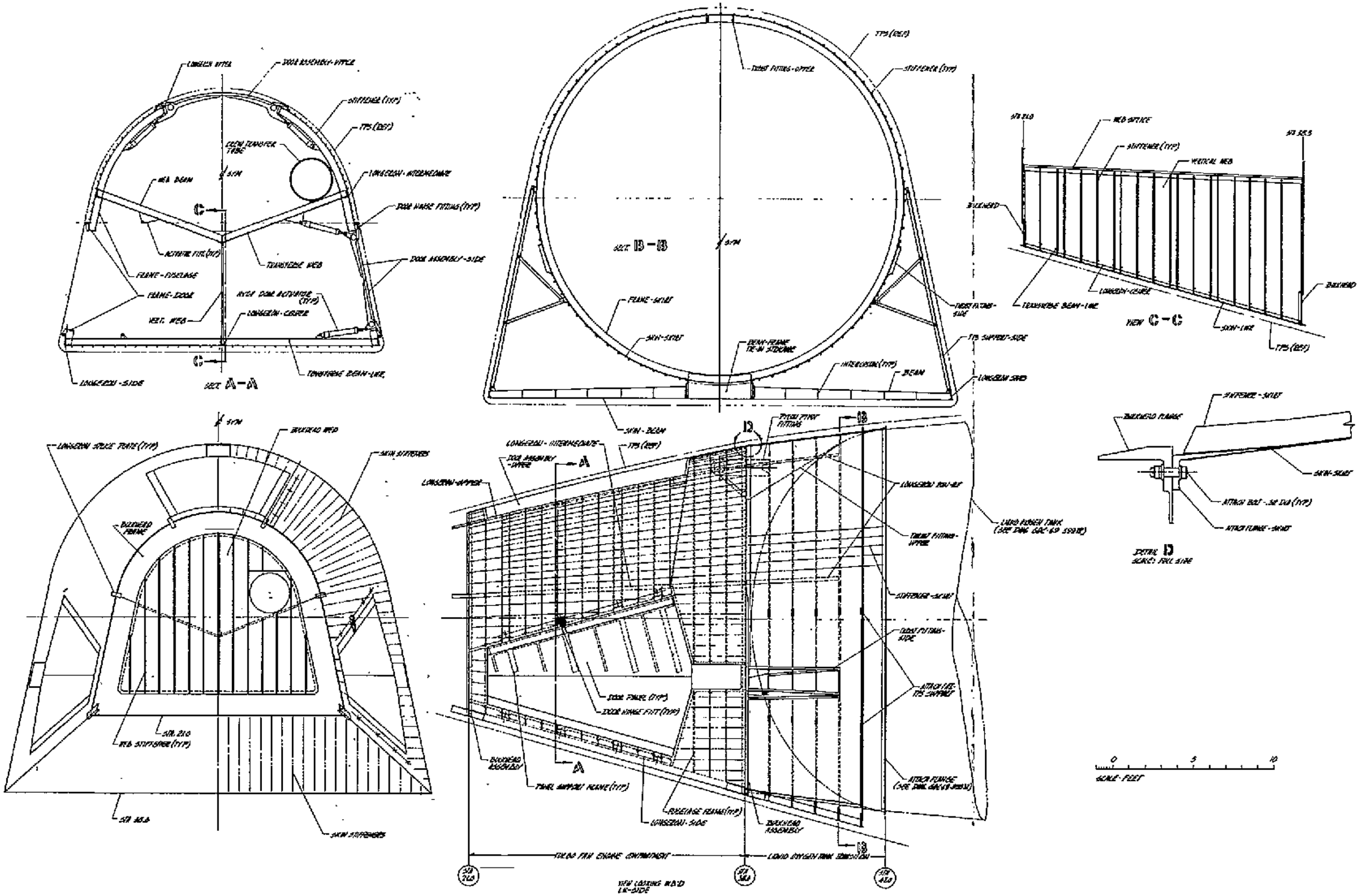

LONGERON UPPER
DOOR ASSEMBLY-UPPER
STIFFENER (TYP)
TPS (REF)
CREW TRANSFER TUBE
SYM
WEB BEAM
LONGERON-INTERMEDIATE
DOOR HINGE FITTING (TYP)
C
ACTUATOR FITG (TYP)
TRANSVERSE WEB
DOOR ASSEMBLY-SIDE
FRAME-FUSELAGE
FRAME-DOOR
HYDR DOOR ACTUATOR (TYP)
VERT. WEB
LONGERON-CENTER
LONGERON-SIDE
TRANSVERSE BEAM-LWR
SECT A-A
TPS (REF)
THRUST FITTING-UPPER
STIFFENER (TYP)
SECT B-B
FRAME-SKIRT
SKIN-SKIRT
BEAM-FRAME TIE-IN STRUCTURE
INTERCOSTAL (TYP)
BEAM
THRUST FITTING-SIDE
TPS SUPPORT-SIDE
LONGERON SIDES
SKIN-BEAM
STA 210
WEB SPLICE
STIFFENER (TYP)
VERTICAL WEB
STA 383
BULKHEAD
TRANSVERSE BEAM-LWR
LONGERON-CENTER
VIEW C-C
SKIN-LWR
TPS (REF)
BULKHEAD
SYM
BULKHEAD WEB
LONGERON SPLICE PLATE (TYP)
SKIN STIFFENERS
BULKHEAD FRAME
STA 210
WEB STIFFENER (TYP)
STA 383
SKIN STIFFENERS
STIFFENER-SKIRT
BULKHEAD FLANGE
SKIN-SKIRT
ATTACH BOLT-.38 DIA (TYP)
ATTACH FLANGE-SKIRT
DETAIL D
SCALE: FULL SIZE
LONGERON-INTERMEDIATE
TPS (REF)
THRUST PYLON FITTING
D
B
DOOR ASSEMBLY-UPPER
A
LONGERON-UPPER
LONGERON RUN-OUT
LIQUID OXYGEN TANK (SEE DWG GDC-69 3991B)
THRUST FITTING-UPPER
STIFFENER-SKIRT
THRUST FITTING-SIDE
DOOR PANEL (TYP)
DOOR HINGE FITT (TYP)
ATTACH FITG-TPS SUPPORT
BULKHEAD ASSEMBLY
A
PANEL SUPPORT FRAME (TYP)
FUSELAGE FRAME (TYP)
LONGERON-SIDE
BULKHEAD ASSEMBLY
ATTACH FLANGE (SEE DWG GDC-69 3991B)
B
TURBO FAN ENGINE COMPARTMENT
LIQUID OXYGEN TANK SECTION
STA 210
STA 383
STA 430
VIEW LOOKING INB'D LH-SIDE
0
5
10
SCALE-FEET

Figure 8-3. Engine Compartment Structural Arrangement

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Each engine bay door assembly consists of a door panel and a panel support frame. The panel is a resistance-welded honeycomb sandwich. Face sheets and core are made of Ti-8Al-1Mo-1V alloy. The support frame, fabricated from the same material, is made up of a number of cantilever beams that are fusion-welded to a four-inch-diameter tube. This tube is as long as the door panel (about 12 feet). It is located near the hinged edge of the door and lends torsional rigidity to the door assembly. The door hinges are bolted to the support frame, in line with a cantilever beam. These beams are provided with lugs for attachment of the hydraulic door actuators.

In general, mechanical fasteners are used to assemble the structure of the turbofan engine compartment. Fusion and resistance welding of structural components shall be considered for subassemblies.

8.4.4 TRANSITION TO LO_2 TANK. The transition skirt provides structural continuity between the bell-shaped cross-section of the forward fuselage and the round LO_2 tank. It consists of a semi-monocoque, stiffened skin shell and a transverse tension field beam. See Figure 8-3.

The shell, or skirt, is a frustrum of a right circular cone, 105 inches long and with an average diameter of 324 inches. It is fabricated entirely from 2024-T6 aluminum alloy. The skin panels of the skirt are contoured sheet metal of medium thickness, except for the positions in the vicinity of the three engine thrust fittings. In these areas, the concentrated engine thrust loads are distributed to acceptable intensity levels at the tank attach flange by a shear lag effect in the thicker skirt skin surrounding each thrust fitting.

At both ends of the skirt, the skin is fastened to attach angles. These angles have a relatively thick and narrow bolt flange, and a longer and tapering skin attach flange. It is stretch-formed from an extrusion to conform to the contour of the skirt skin.

The skirt skin is stabilized by six rings, spaced approximately 18 inches apart. Five of these rings are arranged inside of the shell and attach directly to the skin. The ring nearest the skirt/tank attach flange is fastened to the outside of the skin because the tank dome occupies the space inside of the skirt at this section.

Hat-section stiffeners are arranged around the circumference of the skirt. At both ends, the stiffeners overlap the skin flange of the skirt attach angles. At the aforementioned external ring, the stiffeners pass through cutouts in the ring.

The turbofan engine thrust loads are reacted by and transferred to the skirt skin by fittings that bolt through the Station 38.3 bulkhead to the pylon pivot fittings.

At the bottom of the fuselage section, the TPS support structure has been designed to act also as tension-field beam. The loads in the lower side longerons of the forward fuselage are transferred by this beam toward the centerline. Here, the flanges and the web of the beam are joined to the rings and the skin of the skirt.

The transition structure is bolted to the forware fuselage and the LO_2 tank at the butt-joints formed by its own attach angles and by mating flanges on the fuselage bulkhead and the tank.

8.4.5 LIQUID OXYGEN TANK. The boost vehicle propellant tank structural arrangement is shown in Figure 8-4. The arrangement consists of two integral tanks that are common to both the booster and orbiter vehicles and center-bay auxiliary tanks that are peculiar to the configuration. For the boost vehicle shown in Figure 8-4 the center bay contains one two-cell auxiliary tank. This tank is replaced by seven smaller tanks in the orbiter vehicle as shown in Figure 8-1.

The main LO_2 tank is located between the engine compartment at Station 47.0 and the center body at Station 62.7. It functions as both an integral section of airframe structure and a container for 13,300 ft^3 of LO_2. During operation, the tank is pressurized to a regulated bandwidth of 19.0 to 22.0 psig, with a maximum relief valve setting of 23.0 psig. The tank structure is designed to the 23.0 psig ullage pressure plus hydrostatic pressures with an ultimate safety factor of 1.5. Airframe loads such as shear and bending moment are increased by a safety factor of 1.4 for developing ultimate load conditions.

Thermally, the tank structure is at LO_2 temperature (-320°F) during prelaunch and ascent phases of operation, and is protected from temperatures exceeding 200°F during entry and cruise phases by an outer heat shield. Materials considered for use in tank design were 2219 aluminum in the -T851 or T87 conditions and 718 nickel alloy in the 50 percent cold-rolled condition. As with all of the propellant tanks, 2219 aluminum was selected as the baseline material for illustrating the design concepts. Comparative weight for the tank designed in both materials and for both the booster and orbiter configurations are:

Vehicle	2219 Al	718 Ni Alloy
Booster	8,187 lb	8,981 lb
Orbiter	11,725 lb	11,600 lb

The weight study showed that pressure shell weight is least for 718 material and airframe stability weight is least for 2219 material. With the requirements combined, the total weight trend for this tank is similar for both materials. Final selection of materials will require a more detailed and comprehensive study.

The structural arrangement shown in Figure 8-4 for the LO_2 tank is a skin/stiffener/frame center section with pure monocoque domes on the ends. The compound curvature of the ellipsoidal end domes stiffen the domes adequately for all handling and operational conditions. All joining of the tank shell is by fusion welding, with thickening of the shell structure at the welds to offset reduced properties.

Joining of the tank to the forward and center body structure is shown in Details A and B of Figure 8-4. The butt-ring splice concept shown provides the most easily manufacturable joining of the major airframe assemblies. Stress and cost analysis and prior test experience show that this concept of major assembly joining is structurally reliable and has the lowest weight and cost for the geometry and load intensities encountered in these splice areas.

Access to the tank interior for inspection and maintenance is provided through a removable hatch located at the center of the forward bulkhead. This location aligns with a cutout in the forward fuselage bulkhead located at Station 38.3. The removable door would be similar in concept to the bolted and sealed doors used in similar applications on most present-day missile propellant tanks.

8.4.6 AUXILIARY PROPELLANT TANKS. The auxiliary propellant tanks provide propellant storage in the center body between Stations 72 and 134, where the vehicle interior changes between the booster and orbital configurations. Figure 8-4 presents the boost vehicle auxiliary tank arrangement. The alternative configuration for the orbiter vehicle is shown in Figure 8-1, where the booster auxiliary tank is removed to provide space for the payload and seven smaller tanks are arranged along the center body lower surface to augment the main propellant storage. Commonality of the airframe structure between the two vehicles is maintained by making the airframe structure separate from the auxiliary tanks and designing the tanks for propellant storage only.

The booster auxiliary tank contains 26,300 ft^3 and is divided by an internal bulkhead into two smaller tanks. Storage for 21,900 ft^3 of liquid hydrogen is provided in the forward tank; the aft tank contains 4400 ft^3 of LO_2. During operation, the hydrogen tank is pressurized to a regulated bandwidth of 25.0 to 27.5 psig, with a maximum relief valve setting of 28.5 psig. The hydrogen tank structure is designed to carry the 28.5 psig ullage pressure. The design ullage pressure in the auxiliary LO_2 tank is the same as in the main LO_2 tank (23.0 psig). The auxiliary tanks are also designed to withstand ullage plus hydrostatic pressures with an ultimate safety factor of 1.5. An arbitrary load of 100 lb/in. compression along the longitudinal axis of shell wall was imposed to define a wall-stiffening grid consistant with normal handling and non-operating conditions during which the tank must maintain geometry and structural integrity.

As shown in Figure 8-4, the hydrogen tank is internally insulated. With the outer heat shield protecting the tank from entry heat loads, the operating temperature range for the hydrogen tank structure is between 0 and 200°F. The auxiliary LO_2 tank operates through the same thermal range as the main LO_2 tank (-320 to 200°F). Materials selected for weight estimates are 2219 aluminum and 718 nickel alloy. Comparative weights using each of these materials are:

Vehicle	2219 Al	718 Ni Alloy
Booster	9,995 lb	9,507 lb

As with the main LO_2 tank, the pressure shell portion of the weight is least for 718 material and compression stability weight is least for 2219 aluminum. The tank configuration and design concepts shown in Figure 8-4 are for aluminum material.

As the results of the weight estimate indicate, there appears to be a significant advantage in designing this tank in steel. This is because the pressure vessel weight saving in steel is greater than the weight penalty for the arbitrary shell compression load. Further design work on this concept of tank should consider a steel configuration.

The structural arrangement shown in Figure 8-4 for the booster auxiliary tank is a two-cell, grid-stiffened cylindrical shell closed at the ends and divided by monocoque ellipsoidal domes. The general structural arrangement is further discussed in Volume IV, Section 6, Stress Analysis.

Some design detail of the tank joints and method of support are given in Details C and D of Figure 8-4. Longitudinal support for the tank is at the juncture of the intermediate bulkhead and the cylinder. This arrangement supports both the hydrogen and oxygen propellant loads with all the shell structures in tension and with minimum length of load path. Lateral support occurs at both the forward end (Station 81.0) and the intermediate bulkhead (Station 115.8). Joining of the major elements of the tank structure is by fusion welding at thickened welding lands or at transition rings machined to provide the required transition geometry as well as the welding surfaces.

Access to the tank interiors is provided through removable hatches located on the end domes in accessible and uncongested areas.

Weight estimates for the seven smaller auxiliary tanks used in the orbiter vehicle were made assuming the tank structures to be similar to that shown for the booster auxiliary hydrogen tank. Weight was estimated for an aluminum configuration only:

Tank	Weight
9 by 40 ft LH_2 Tanks (2)	2000
8.5 by 62 ft LH_2 Tank (1)	1700
6.7 by 23 ft LH_2 Tanks (2)	840
4 by 11.5 ft LO_2 Tanks (2)	300
Total Weight	4840 lb

8.4.7 LIQUID HYDROGEN TANK. The main LH_2 tank is located between the center body at Station 143.3 and the rocket engine thrust section at Station 174.7. It functions as both an integral section of airframe and a container for 24,400 ft^3 of LH_2. During operation, the tank is pressurized to a regulated bandwidth of 25.0 to 27.5 psig, with a maximum relief valve setting of 28.5 psig. As with the LO_2 tank, the structure is designed to the maximum ullage pressure (28.5 psig) plus hydrostatic pressure with an ultimate safety factor of 1.5. An ultimate safety factor of 1.4 is used for designing the primary airframe part of the tank structure.

As shown in Figure 8-4, the LH_2 tank is internally insulated; with the outer surface heat shield, the structure will operate through a temperature range of 0 to 200°F. Materials considered for the tank structure were the same as for the other tanks: 2219 aluminum and 718 nickel alloy. Estimated weights for these materials for both orbiter and boost vehicles are:

Vehicle	2219 Al	718 Ni Alloy
Booster	13,122 lb	15,981 lb
Orbiter	13,541 lb	14,403 lb

As previously discussed, a common tank is lighter in aluminum and this is the material considered in the design concept shown in Figure 8-4.

The structural arrangement shown for the hydrogen tank is a frame/stringer stiffened-center cylindrical section with monocoque ellipsoidal end domes. This arrangement is further discussed in Volume IV, Section 6, Stress Analysis. The general concepts of fabrication and accessibility to the tank interior are the same as those previously discussed for the other tanks. Differences in the design concept occur at the frames and at the thrust structure splice.

The frames are located external to the tank shell to provide a smooth inner surface for insulating. Attachment of the frames to the grid-stiffened tank shell is by mechanical fasteners through circumferential rings machined into the shell grid. Depth of the frames varies from 4 inches across the top of the vehicle to 12 inches along the lower quadrants. The greater depth across the bottom provides the frame strength and stiffness needed to distribute the lower surface entry airload pressures into body shear along the tank shell.

A potential problem in the frame design occurs across the top quadrants, where the spacing between the frame outer flange and the fuselage outer surface heat shield is about two inches. This stand-off distance approaches a minimum acceptable dimension and may cause local hot spots on the frame flanges. If more detailed analysis shows this heating to be too high for aluminum, the frames could be made from titanium or 718. The bi-metallic influence (titanium frame and aluminum shell) on both thermal and pressure discontinuity effects can be compensated for in the local design of the frame/shell attachment. Since the stability requirement for the frame in this area is constant EI, frame cap area can be reduced proportionately as E increases.

Joining of the thrust section to the hydrogen tank is shown in detail in Figure 8-4. A shear splice is shown for this joint because of the magnitude and reversible nature of the loads in this area. The splice ring is separately machined and welded into the tank structure. Away from zones of high thrust-section loads, the ring could be lightened by pocket milling excess material as shown in similar rings at Details B and E. The blade extending aft provides for double shear attachment of the thrust structure splice plates.

8.4.8 CENTERBODY. The entire centerbody, which includes all primary structure between the LH_2 forward bulkhead and the LO_2 aft bulkhead, is discussed in this section. (See Figure 8-2.)

Centerbody Structure

Geometry of the centerbody structure is dictated by the payload requirements. The 62-foot distance between the LH_2 and LO_2 bulkheads resulted from the groundrule of a 15-foot-diameter by 60-foot-long payload envelope with one foot of clearance all around. The centerbody structure provides for support and transmission of all loads imposed in the area between the integral structure LO_2 and LH_2 propellant tanks.

The forward and aft landing gear and the variable-geometry wing with all the associated mechanisms and support hardware are also supported by the centerbody structure. The primary body structure is generally circular from its attachment to the LH_2 tank forward to the wing bulkhead. At the wing bulkhead, the structure transitions to a pear shape to accommodate maximum space utilization for supplementary propellant tanks in the orbiter. The pear-shaped body structure extends approximately 25 feet forward from the wing bulkhead. In the 8-foot distance between the forward landing gear drag strut bulkhead and the forward landing gear bulkhead, the structure transitions back to a circular shape. The circular shape continues forward to the interface with the LO_2 tank.

Parametric studies indicate that aluminum alloy skin/stringer structure provides the best combination of weight, economy, and reliability. Loads generated during this study indicate that an average equivalent skin gage slightly less than 1/8 inch is required. This equivalent gage in aluminum alloy is based on a configuration effectivity of 80 percent and a material efficiency of 67 percent. A one-foot radial envelope was allocated around the perimeter of primary structure for skin, stringers, and frames.

A number of configurations of the centerbody to integral structural propellant tank joint have been investigated. (See Figure 8-5.) Preliminary results indicate that the best combination of weight, economy, and fabricatability is the tension joint located away from the tangent point of the elliptical bulkhead.

The wing bulkhead consists of a number of structural elements, as indicated in Figure 8-6. A 35-foot-long, 4-foot-deep, and 3-foot-wide truss extends across the base of the vehicle. At either end of the truss is a machined clevis fitting for attachment of the variable-geometry wing. This truss is designed to carry through the centerbody the substantial bending moment of the wing. For this reason, Al/boron composite was selected for use on the truss beam caps. It is intended that the truss and clevises be machined from titanium 8-1-1.

The wing bulkhead side members are shaped to accommodate the orbiter supplementary tanks and to provide a gusseted transition between the circular body structure aft and the pear-shaped body structure forward. These side members are three feet deep to conform to the size of the base truss, and extend around the periphery to a radial position 12 degrees above the body circular section centerline. At this location, two standard frame sections splice into each side of the wing bulkhead. It is at this splice that a longeron extending 10 feet forward and aft of the wing bulkhead provides for continuity of the centerbody structure through its transition in shape. A considerable torsional load and a limited drag load are reacted from the wing through the wing bulkhead to the body structure by a 40-inch-deep, 180-inch-long beam which extends forward and upward from the bulkhead clevis to the heavy bulkhead provided for the interstage fitting drag strut. The beam caps are fabricated from a high strength composite and are tapered to the beam web width at their forward end. The beam web thickness is constant because of the constant shear condition and is attached both forward and aft.

The stage interconnect drag strut support bulkhead is a machine-sculptured structure. A web nearly tangent to the side-body skin/stringer picks up the torsion/drag beam from the wing bulkhead and thus provides a good load path for shearing out the kick load. In addition, this bulkhead has flange provisions to carry-through the continuity of the longitudinal loads in the body skin/stringer structure.

The landing gear/payload compartment closure bulkhead for both the forward and aft configurations of this bulkhead are continuous, as shown by Sections A-A and D-D in Figure 8-2. The primary loads reacted result from the landing gear attachment and the lateral loads from the payload doors and payload support longeron. At this location between the payload envelope and the propellant tank bulkheads, however, there is the capability to provide lateral stability in the plane of the bulkhead, thus reducing the complexity of load paths in this bulkhead and allowing for the use of aluminum alloy for its fabrication.

As shown in Figure 8-7, the payload bay door does not carry primary flight loads. It is designed to accommodate a purge differential pressure condition, a ground wind condition, and flight reentry thermal condition. Temperatures in the leeward area where the door is presently located allow for the design of a hot-structure titanium (Ti 8-1-1) door. The configuration of this point design uses a piano-type hinge to mount the door flush with the outside contour TPS. Each of the two doors covers

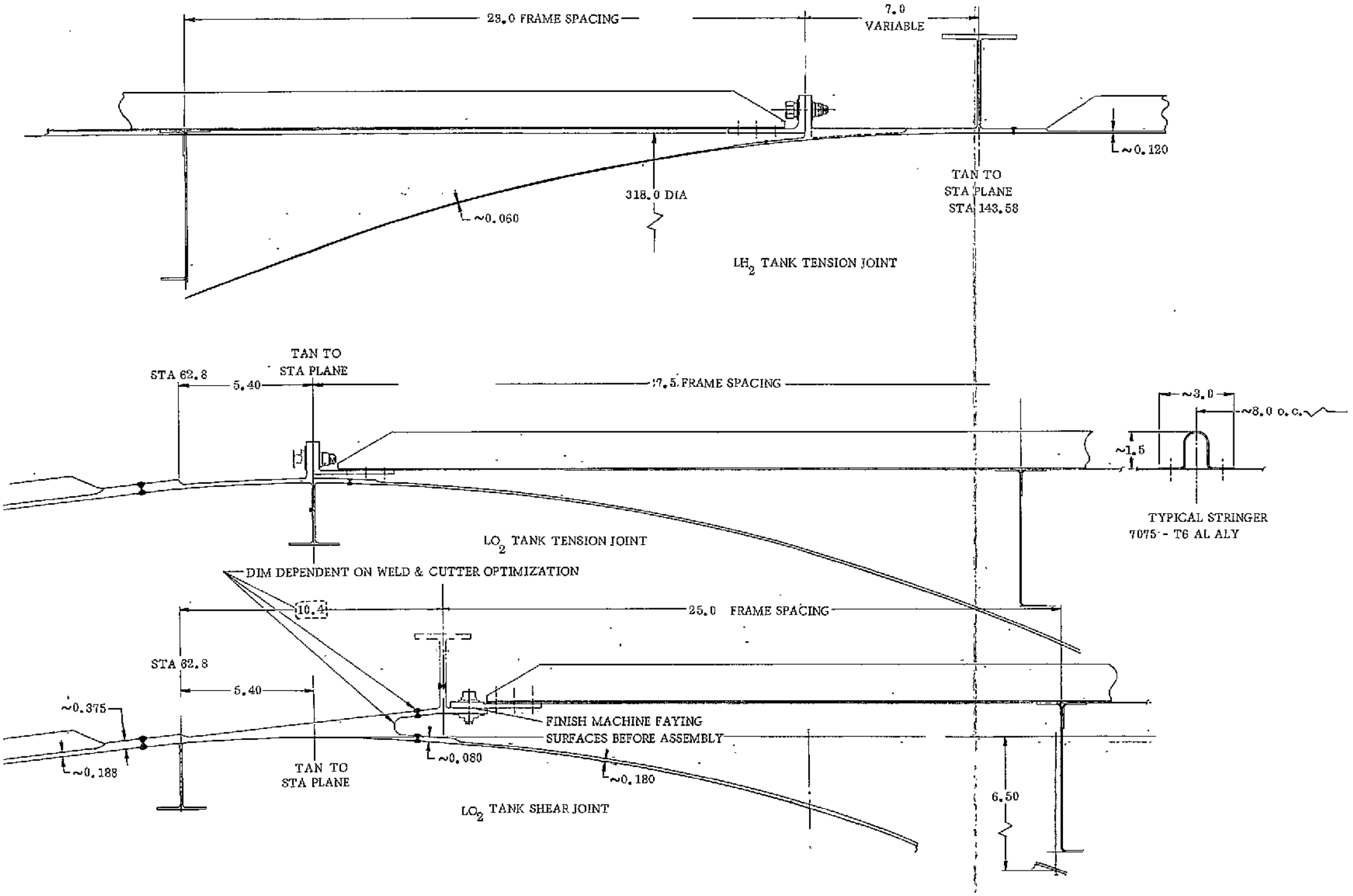

23.0 FRAME SPACING
7.0
VARIABLE
~0.120
TAN TO
STA PLANE
STA 143.58
318.0 DIA
~0.060
LH_2 TANK TENSION JOINT
STA 62.8
TAN TO
STA PLANE
5.40
17.5 FRAME SPACING
~3.0
~8.0 o.c.
~1.5
TYPICAL STRINGER
7075 - T6 AL ALY
LO_2 TANK TENSION JOINT
DIM DEPENDENT ON WELD & CUTTER OPTIMIZATION
10.4
25.0 FRAME SPACING
STA 62.8
5.40
~0.375
~0.188
TAN TO
STA PLANE
~0.080
FINISH MACHINE FAYING
SURFACES BEFORE ASSEMBLY
~0.180
LO_2 TANK SHEAR JOINT
6.50

Figure 8-5. Attachment of Centerbody to Propellant Tank

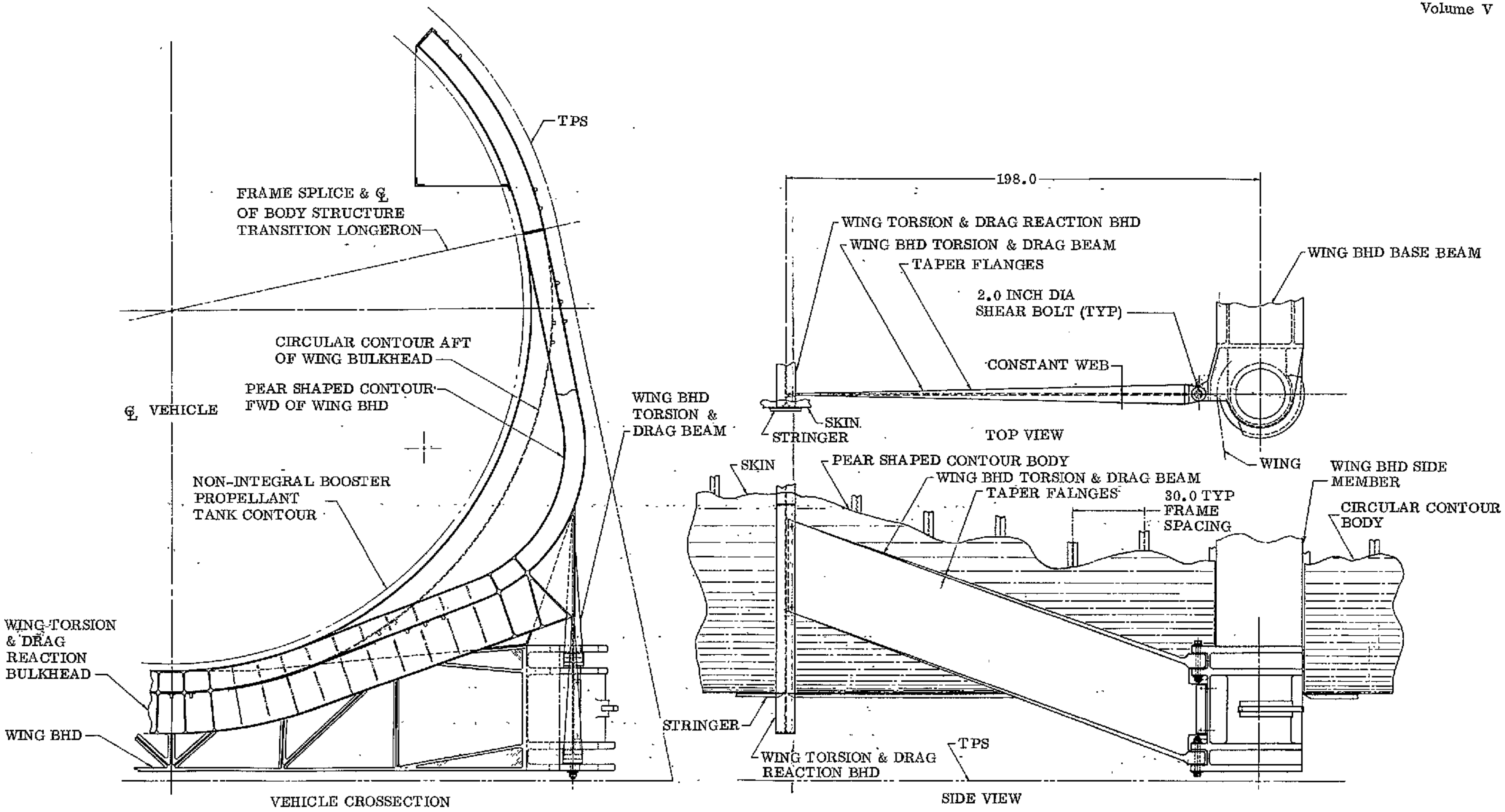
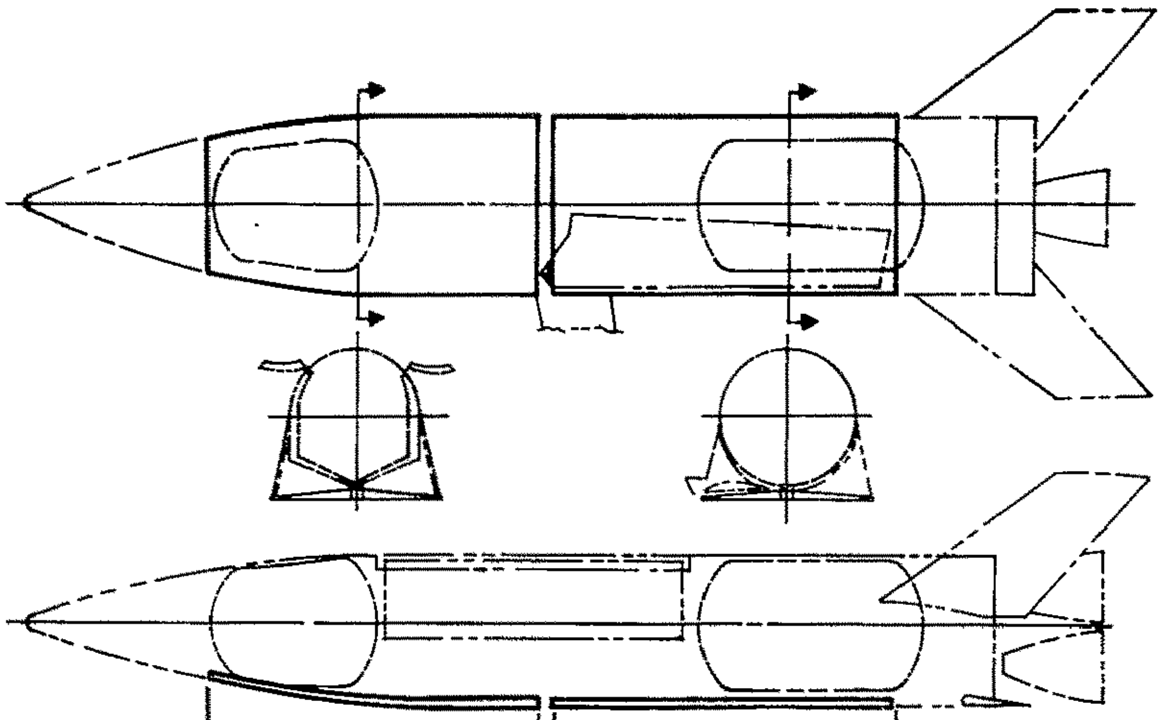
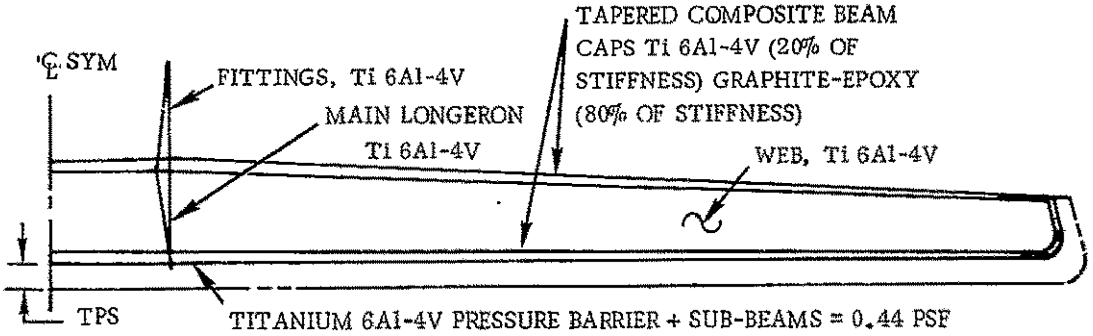

TPS
FRAME SPLICE & ℄
OF BODY STRUCTURE
TRANSITION LONGERON
CIRCULAR CONTOUR AFT
OF WING BULKHEAD
PEAR SHAPED CONTOUR
FWD OF WING BHD
℄ VEHICLE
WING BHD
TORSION &
DRAG BEAM
NON-INTEGRAL BOOSTER
PROPELLANT
TANK CONTOUR
WING TORSION
& DRAG
REACTION
BULKHEAD
WING BHD
VEHICLE CROSSECTION
198.0
WING TORSION & DRAG REACTION BHD
WING BHD TORSION & DRAG BEAM
TAPER FLANGES
2.0 INCH DIA
SHEAR BOLT (TYP)
WING BHD BASE BEAM
CONSTANT WEB
SKIN
STRINGER
TOP VIEW
WING
SKIN
PEAR SHAPED CONTOUR BODY
WING BHD TORSION & DRAG BEAM
TAPER FALNGES
30.0 TYP
FRAME
SPACING
WING BHD SIDE
MEMBER
CIRCULAR CONTOUR
BODY
STRINGER
WING TORSION & DRAG
REACTION BHD
TPS
SIDE VIEW

Figure 8-6. Wing Support Structure



0.89 PSF FOR TPS SUPPORT STRUCTURE AND PRESSURE BARRIER (0.45 PSF MAIN BEAMS + 0.44 PSF SUB-BEAMS AND PRESSURE BARRIER)

1.04 PSF FOR TPS SUPPORT STRUCTURE AND PRESSURE BARRIER (0.60 PSF MAIN BEAMS + 0.44 PSF SUB-BEAMS AND PRESSURE BARRIER)


℄ SYM
FITTINGS, Ti 6Al-4V
MAIN LONGERON Ti 6Al-4V
TAPERED COMPOSITE BEAM CAPS Ti 6Al-4V (20% OF STIFFNESS) GRAPHITE-EPOXY (80% OF STIFFNESS)
WEB, Ti 6Al-4V
TPS
TITANIUM 6Al-4V PRESSURE BARRIER + SUB-BEAMS = 0.44 PSF

DETAIL OF MAIN TPS SUPPORT BEAM UNDER WING (2.50 FT SPACING)

UNIT WEIGHT: 0.60 PSF FOR TPS SUPPORT BEAMS UNDER WINGS (CANTILEVERED FROM CENTER)

UNIT WEIGHT: 0.45 PSF FOR TPS SUPPORT BEAMS FORWARD OF WINGS (OUTER ENDS OF BEAMS BRACED BY STRUTS)

Figure 8-20. Base TPS Support Structure

Base Heat Shield Support

The base heat shield support is similar to Saturn SI-C and consists of 26 honeycomb panels approximately three by five feet and one inch thick, with a hot-layer side of insulating material 0.4 inch thick. The heat shield supporting structure is located in the plane of the aft thrust ring at Station 189.5 and is supported by the thrust structure and auxiliary vertical and horizontal beams. The vertical beams also serve as compression struts to restrain vehicle separation loads.

Openings through the heat shield for fluid lines are protected by a flexible curtain of fiberglass cloth sandwiched between Inconel wire mesh. Deflection of the shield and engine support beams is limited to two inches, and the differential base pressure is 2.7 psi. The cold-side temperature does not exceed 300°F.

8.4.10 TPS SUPPORT STRUCTURE (FR-3 AND FR-4 ORBITERS AND BOOSTERS). The TPS is supported by structure extending from the basic structure, by mounting on fittings attached to the basic structure, or by both.

The support structure for the basic TPS from the forward end of the LO_2 tank to the aft end of the wing compartments is shown in Figure 8-20.

The main support beams aft of the wing pivot are supported near the centerline of the vehicle. The cantilever length is approximately 16 feet on the orbiters and approximately 18 feet on the boosters. To keep the end deflection to the required one inch under 1 g planform loading, composite materials are used to obtain high stiffness with low weight.

The main support beams are spaced 30 inches apart. The basic structure of these beams is 6Al-4V titanium. The beam caps are long, almost straight, and gradually tapered so it is relatively easy to attach a composite of unidirectional graphite fibers in an epoxy matrix. Stiffness-to-weight ratio of the composite is much higher than for the titanium. The weight saving in the beam caps is about 60 percent.

The base TPS support beams forward of the wings have their outer ends supported by struts extending down from the basic structure. The unit weight of these beams including the struts is therefore lower than for the unsupported cantilever beams aft of the wings.

Intermediate or sub beams between the main TPS support beams on the base are added to provide for mounting heat shield support posts on a grid approximately 15 × 18 inches. These intermediate beams also help support the titanium sheet used as a pressure barrier on the cold side of the insulation.

Lower TPS and Trailing Edge Support Structure

The lower or windward TPS supporting structure and trailing edge support structure, shown in Figure 8-19, is a 35-foot-wide, 27.8-foot-long box structure located below the thrust skirt. The structure consists of two 30-inch-deep main transverse beams positioned in the plane of and below the fore and aft thrust rings at Stations 179.5 and 189.5, respectively. Eleven longitudinal beams support the lower TPS panels and provide moment restraint for the trailing edge overhang. Four main long beams at BL 6.5 and BL 15.5 provide longitudinal shear attachment to the thrust skirt through diagonal panels. Trailing-edge moment restraint is provided by panel strut attachments to the thrust rings.

Vertical and transverse shear load restraint is provided by fore and aft stiffened panels at Stations 179.5 and 189.5 that attach the transverse beams and thrust rings.

A horizontally positioned panel stiffened with transverse formers on the upper side covers the entire TPS and trailing-edge support structure. The forward portion between Stations 179.5 and 189.5 forms a beam that provides horizontal shear load and moment restraint from the holddown fittings to the thrust skirt. The aft portion is attached on the upper side of the trailing edge structure to restrain the positive engine pressure during boost phase. TPS panels are attached to the lower surface of the supporting frames and beams. The shear webs and attaching stiffeners on the beams are fabricated from Ti 8-1-1 material with aluminum-boron composite bonded caps.

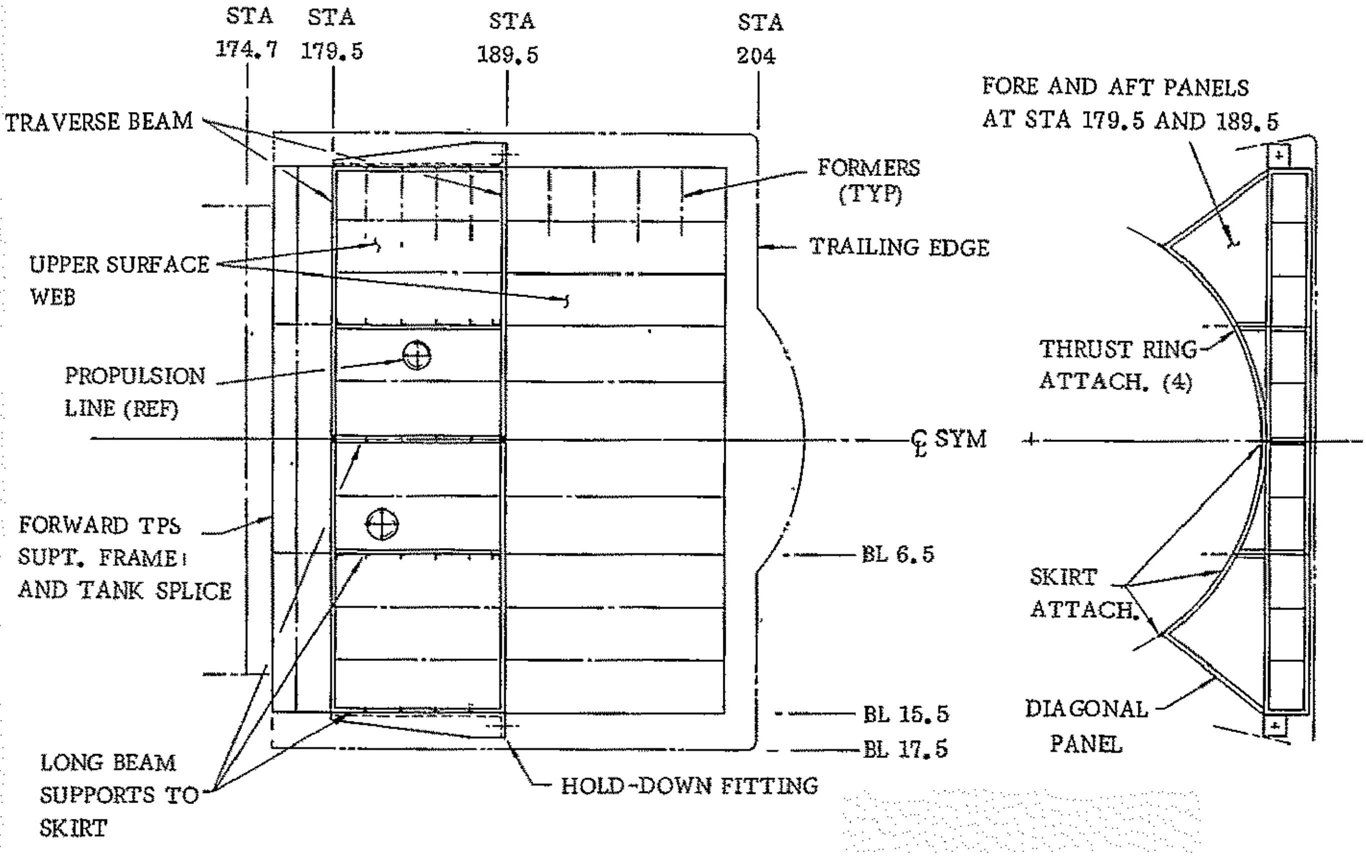

STA 174.7
STA 179.5
STA 189.5
STA 204
TRAVERSE BEAM
FORMERS (TYP)
FORE AND AFT PANELS AT STA 179.5 AND 189.5
TRAILING EDGE
UPPER SURFACE WEB
PROPULSION LINE (REF)
THRUST RING ATTACH. (4)
℄ SYM
FORWARD TPS SUPT. FRAME AND TANK SPLICE
BL 6.5
SKIRT ATTACH.
BL 15.5
BL 17.5
DIAGONAL PANEL
LONG BEAM SUPPORTS TO SKIRT
HOLD-DOWN FITTING

Figure 8-19. Aft Body TPS and Trailing Edge Support Structure

The forward and aft spars of the torque box structure each contain two-lugged fittings that project through the thrust skirt. These fittings attach to mating fittings in each stabilizer through single-bolt attachments that provide moment restraint to the fore and aft spars of the stabilizers. Vertical restraint of both stabilizers is provided through the rear spar attachments, which are supported by the aft thrust ring.

The forward thrust ring in the plane of the upper engine support beam contains a one-lug fitting that projects through the skirt. This fitting provides drag and vertical load restraint of the stabilizer through a one-bolt attachment to the forward stabilizer spar. Drag loads are distributed through the skirt by two longerons attached through the skirt to the inner beam gussets.

Vehicle Separation Support Structure

Two vehicle separation restraint fittings (Figure 8-12) are located at both the upper and lower surfaces of the vehicle at BL 6.5 and in the plane of the aft thrust ring at vehicle Station 189.5. These fittings provide vertical and horizontal load restraint of the interconnecting linkage to the adjacent boost vehicle during the boost phase of flight. The vehicle interconnecting linkage is laterally positioned to clear the main propulsion lines to the engines. The upper fittings are attached directly to the thrust skirt and project through the upper TPS surface of the vehicle. The fittings are titanium structures 3.5 feet high and 10 feet long, and are attached through the skirt to the forward and aft thrust rings.

Transverse load restraint is provided by two vertical struts (wide flange beams) of titanium Ti 8-1-1 and aluminum-boron cap material. Each strut is 29 feet long, but is fabricated in three sections to accommodate the lateral engine thrust beams. Strut-cap continuity is provided by splice plates that span the crossbeams.

These members also support the base heat shield against engine thrust pressure. Both struts are bolted to the web of the aft thrust ring and project through the thrust skirt. The lower end of each strut is also bolted to the web of the trailing edge support beam and terminates at the lower surface of the vehicle at the vehicle interconnect fittings.

Longitudinal loads are restrained by long beams at BL 6.5, which are attached to the struts at Station 189.5 and run forward to Station 179.5. Longitudinal loads are sheared inward to the supporting thrust skirt, and the resulting moment is restrained by the fore and aft thrust rings. These long beams, with the addition of the outboard long beams that support the lower holddown fittings, form the major supporting structure for the intermediate frames of the lower TPS and trailing edge supporting structure.

Stabilizer Center Section

The stabilizer center section, shown in Figures 8-12, 8-16, and 8-18, consists of a closed titanium Ti 8-1-1 torque box structure which is 4 feet deep, 5 feet high, and 26 feet long.

The upper engine-support beam, which projects through the skirt, forms the upper surface of the torque box structure. The fore and aft beams forming the side of the box structure vary in depth from 5 feet at the center to 3.8 feet at the outboard ends. This offset provides for the three-foot-diameter inlet LO_2 line to the upper engine. The lower surface of the torque box structure is formed of stiffened panels supported at the point of beam convergence by four vertically positioned internal bulkheads. The change in lateral beam depth of the torque box at BL 6.5 coincides with the vertically positioned vehicle separation support struts. Closure bulkheads located on the outboard ends of the torque box provide structural continuity and support for the stabilizer attachment fittings.

The torque box extends through a rectangular cutout in each side of the thrust skirt. Match angles with mechanical fasteners are used to attach the torque box structure to the thrust skirt structure.

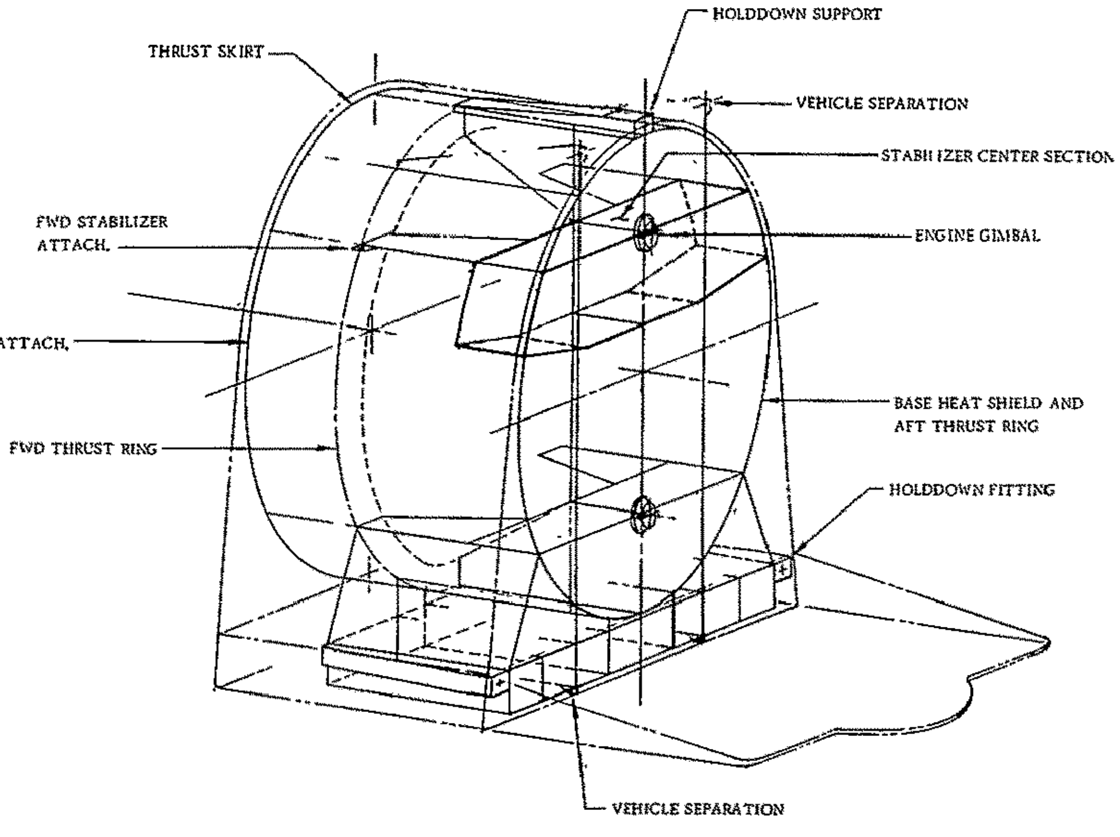

HOLDDOWN SUPPORT
THRUST SKIRT
VEHICLE SEPARATION
STABILIZER CENTER SECTION
FWD STABILIZER ATTACH.
ENGINE GIMBAL
LH2 TANK "Y" RING ATTACH.
BASE HEAT SHIELD AND AFT THRUST RING
FWD THRUST RING
HOLDDOWN FITTING
VEHICLE SEPARATION

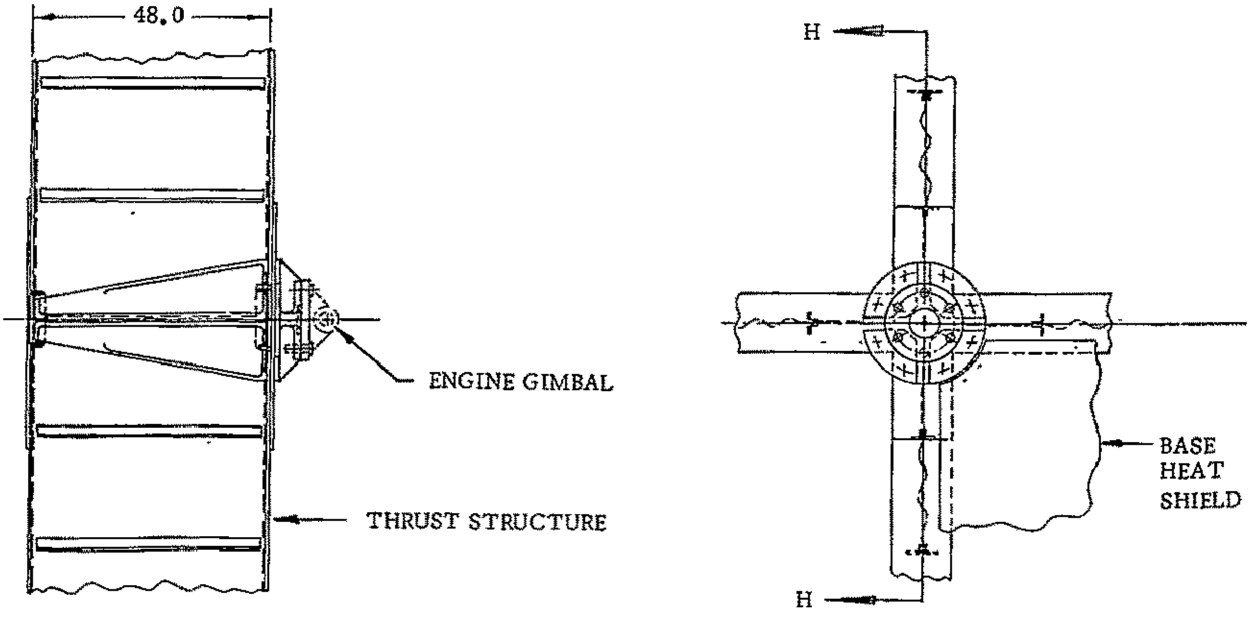
Figure 8-18. Stabilizer Center Section

The upper and lower flanges of the two transverse crossbeams are continuous from end to end (22 feet). The vertically positioned beam consists of three beam subassemblies that are welded to opposite sides of the transverse beams at the vehicle centerline. Flange continuity is achieved by welding and doubler assemblies. Additional rigidity in the lateral beams is provided by the use of unidirectional aluminum-boron composite caps bonded to the titanium weldment. The material is proportioned to provide equal beam restraint at each beam/skirt attachment and a more uniform load distribution in the thrust skirt.

The main flanges of the engine support beams are tapered titanium bars and incorporate a pair of continuous small flanges that are diffusion-bonded to the flange material. These tabs match the material of the auxiliary flange of the web assembly in thickness and are welded to the main flanges to form a beam.

The outboard ends of the crossbeam web terminates in a T section. The stem of the T extends outboard and is bolted to a mating doubler and web of the thrust skirt. This mechanical joint provides for radial adjustment in the crossbeam assembly.

The fitting attaching the engine gimbal to the support structure shown in Figure 8-17 is a separate titanium fitting that is bolted, through the lower flanges of the crossbeams, to the cruciform section post. The gimbal attachment fitting is fabricated from titanium plate and is of welded construction; it is basically a tubular configuration. The closure plate at the lower end accepts the attachment of the gimbal fitting on the engine, and a flange at the upper end of the fitting is used to bolt it to the support structure.



SECTION H-H

VIEW G-G

Figure 8-17. Engine Gimbal Support Fitting

The webs of the crossbeams are fabricated from corrugated titanium sheet materials, continuously welded to an auxiliary flange of flat sheet material. Corrugated webs are integrally stiffened by their form and are efficient shear members with continuous flange support.

Holddown Posts

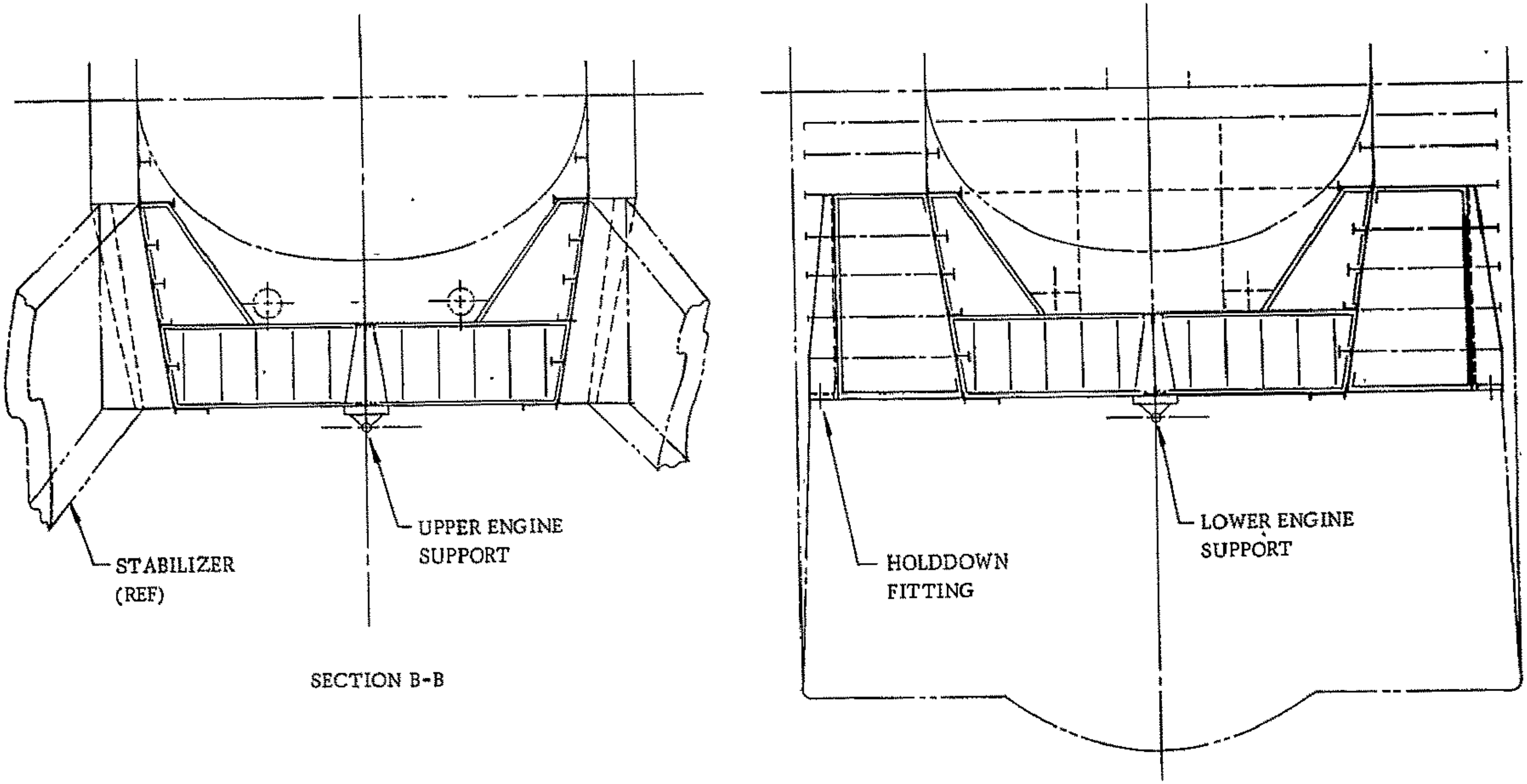
Three holddown posts on the periphery of the thrust structure support the entire weight of the vehicle when it is standing on the pad, provide moment restraint for wind loads during launch, and introduce the launch rebound loads (which occur when thrust is abruptly terminated on the pad) into the shell structure. The fittings shown in Figure 8-12 are 10-foot-long Ti 8-1-1 titanium alloy weldments with a two-inch-thick, 12-inch-square pad at the base, which mates with the launch gear at vehicle Station 189.5. The upper fitting is bolted directly to the thrust skirt and restrained by the forward and aft thrust rings and the vertical thrust beam assembly. A tapered titanium doubler, attached to the inner surface of the thrust skirt, is used to distribute the concentrated load from the upper holddown fitting to the thrust skirt.

Each of the two lower fittings is attached to the outer long beams that form the outer periphery of the transverse frame structure which supports the lower TPS panels.

The lower holddown fittings are also restrained by two long beams that span between vehicle Stations 179.5 and 189.5. The diagonal beams are 7.5 feet deep and are stiffened in the station plane of the thrust skirt frames at two-foot intervals. The inboard side is fastened through the thrust skirt to the lower thrust beam. Vehicle holddown loads are sheared across the diagonal beams to the skirt. Kick loads generated by the offset holddown fittings are restrained by the fore and aft thrust rings.

A horizontal panel, which spans the upper surface of the lower TPS supporting frames and extends from vehicle Stations 179.5 to 189.5, also acts as part of the redundant structure that supports the lower holddown fittings. The panel, which is 31 feet wide and spans between holddown fittings, is stiffened by both lateral and longitudinal frames at two-foot spacing. The outer edges of the panel are attached to the upper edge of each long beam and the diagonal beams. The panel is also attached to the thrust skirt along the vehicle centerline. All holddown-fitting supporting structure is fabricated from titanium alloy sheet and angle stiffeners.

The lower holddown fittings are also used in raising the vehicle to a vertical position on the pad. Two of the three ground-support arms will contain in-line pivot fittings below the holddown clamps. These arms are clamped to the lower holddown posts on the base of the vehicle and allow vehicle rotation about its base during the erection procedure.


STABILIZER (REF)
UPPER ENGINE SUPPORT
HOLDDOWN FITTING
LOWER ENGINE SUPPORT

SECTION B-B

SECTION C-C

Figure 8-16. Thrust Beam/Stabilizer Support Structure

The 20-inch-deep frame uses a continuous-weld, radially corrugated web with the flanges attached to the web through an auxiliary flange of sheet titanium. The web incorporates local flat areas where the forward ends of the thrust and holddown posts are bolted to the web.

The frame is fabricated in quarter-section assemblies that are assembled into a complete frame.

Aft Thrust Ring

The frame shown in Figure 8-12 at vehicle Station 189.5 terminates the aft end of the shell structure and is a highly-loaded structural element. Major loads on the frame result from engine thrust and gimbal loads that are introduced into the shell structure.

The 18-inch-deep frame uses a continuous-weld, corrugated web construction. Since corrugations in the frame web are generated by radial lines, the corrugated sheet is not of constant cross-section. Corrugated pitch and amplitude are greater at the outboard edge of the frame than at the inboard edge. Splices between corrugated sheets are therefore accomplished on straight radial lines.

A circular frame loaded in bending subjects the web to significant crushing or tension loads. The web is loaded in radial compression when the outer flange is loaded in compression. The corrugated web is well suited to the crushing loads that are encountered in the frame web.

Flat web sections are provided at the thrust beams, holddown post attachment fittings, stabilizer rear spar carry-through, and vehicle separation support beams. Major fittings are attached by mechanical fasteners for producibility considerations.

The lower (aft) end of the holddown post is bolted through the thrust ring cap and support fitting attached to the frame web.

The main flanges of the frame are machine-tapered titanium bars and incorporate continuous small flanges for welding the web subassemblies to the main flanges.

Engine Support Beams

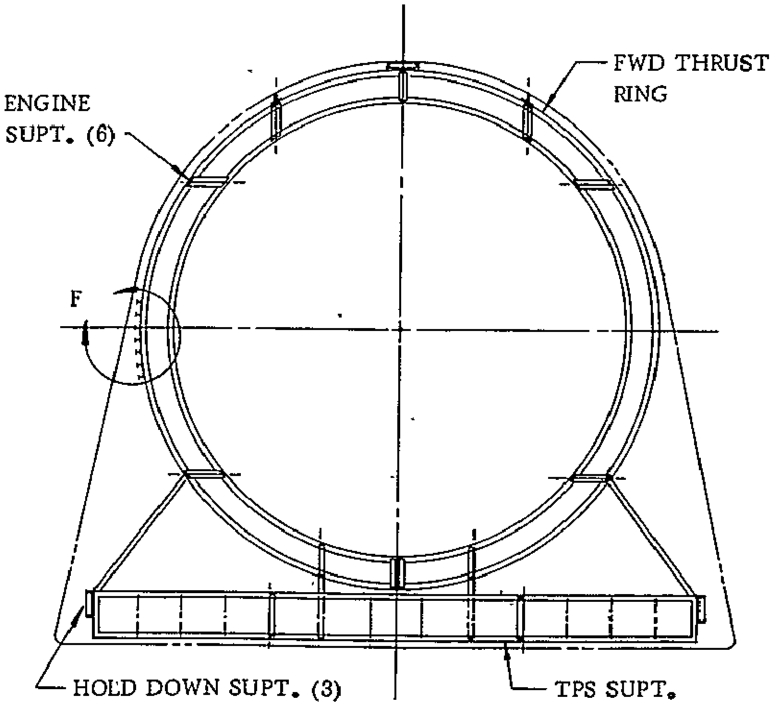
The thrust structure, shown in Figures 8-12 and 8-16, consists of two horizontal four-foot-deep beams that intersect a centrally positioned vertical beam at the two engine gimbal support fittings. Beams are mutually perpendicular and span across the cylindrical thrust structure. The aft surface of the beam matrix at vehicle Station 189.5 is attached to the aft thrust ring and is used to support the base heat shield panels. The ends of each beam are attached to the skirt and are reinforced by large gussets extending to the forward thrust ring at vehicle Station 179.5. The beams and beam gussets are sized to provide clearance for the engine inlet lines and engine gimballing.

Intermediate Frames

Four intermediate frames, shown in Figures 8-13 and 8-14, are located between the major fore and aft frames located at vehicle Stations 189.5 and 179.5, respectively. These six-inch-deep frames are of corrugated-web welded construction. At each station, the frame is made up of six segments; each segment spans between adjacent thrust beams and is approximately 30.0 inches wide at its center and 165 inches long. Ends of the corrugated web segments terminate with a cruciform extrusion that is mechanically attached to a machined fitting. The end fitting is bolted to the thrust beams.

Forward Thrust Ring

The circular frame shown in Figure 8-15 at vehicle Station 179.5 is a major structural frame. The forward ends of the thrust posts and holddown posts are bolted to the web of the frame. Major loads on the frame are radial loads arising from axial loads in the posts being resisted at the vehicle skin line. The moment resulting from this offset is resisted by couple loads at vehicle Stations 179.5 and 189.5. The forward thrust ring also restrains kick loads from the vehicle separation restraint fittings and loads from the forward spar of the stabilizer.



SECTION E-E

Figure 8-15. Forward Thrust Ring

An alternative thrust skirt concept employs a boat-tail configuration that provides a 1,450-pound structural weight saving. This configuration is formed by a change in shape aft of the circular forward thrust ring at Station 179.5. The transition employs straight-line elements to an elliptical aft thrust ring at Station 189.5. Both lateral thrust-support beams are therefore shortened three feet. Additional cost is imposed, since each of the four intermediate frames in the transmition area has a different elliptical shape.

Thrust Structure - Tank Joint

The titanium thrust structure is terminated at its upper (forward) end at vehicle Station 176.2 with a shear-type attachment (Figure 8-14).

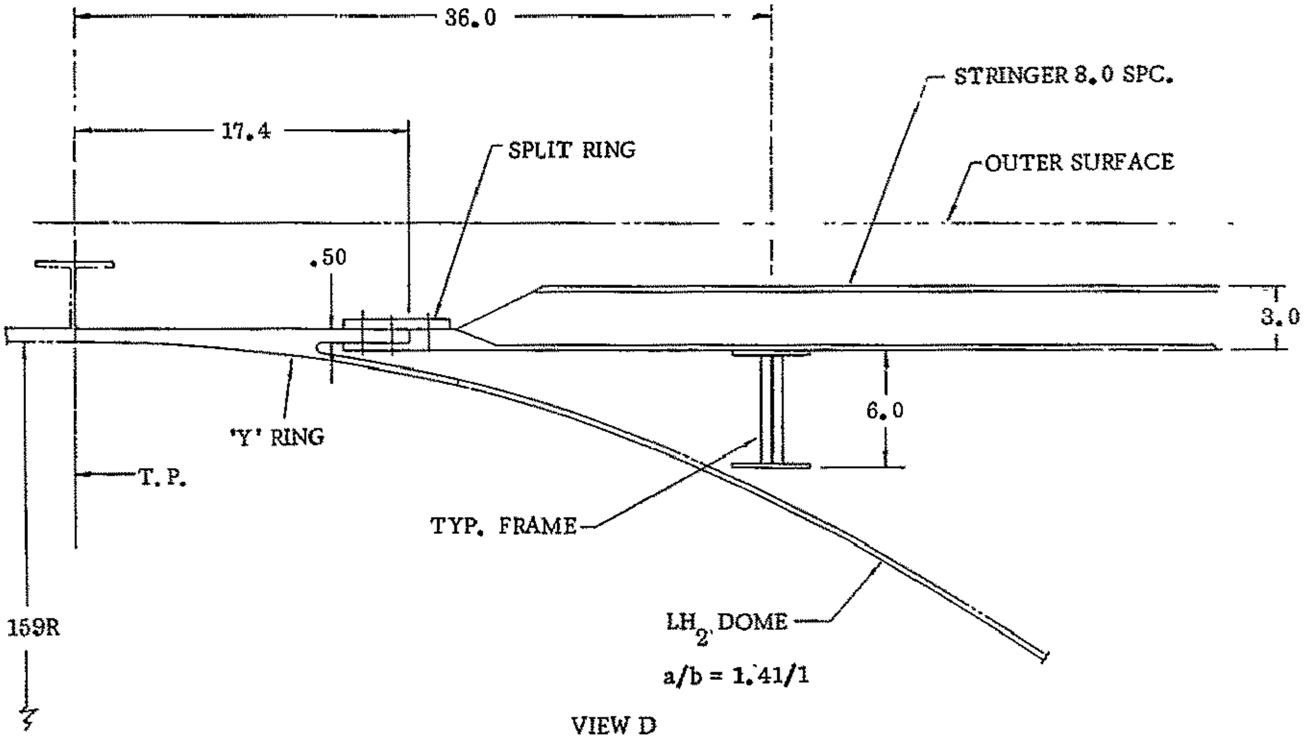


Figure 8-14. Thrust Skirt Attachment Shear Splice

The major forces on the structural splice load the joint in compression (about 3,000 pounds/inch). With the addition of an outer split-ring, all loads are transferred across the bolted joint in double shear. At the forward end of the thrust structure, a stepped lip is machined in the titanium alloy skirt structure to mate with the 2219-T87 aluminum alloy tank Y ring. Shims of 0.03 maximum thickness are provided to ensure proper fit at installation.

In the area of the upper holddown fitting, the running load through the skirt attachment is increased since the short skirt length does not allow the concentrated load to dissipate sufficiently in the skirt. The larger load on the upper side is therefore restrained by the tank wall through an increased wall thickness at the splice.

Skin/Stringer

The thrust skirt is composed of skin panels approximately 4 feet wide and 13.5 feet long. Three or four integrally stiffened skin panels are joined by longitudinal weld seams into larger curved panels. Each panel is composed of machined titanium plate of varying skin thickness with 0.5-inch outstanding legs spaced at five-inch intervals. Machined T sections are attached to the outstanding legs by fusion welding.

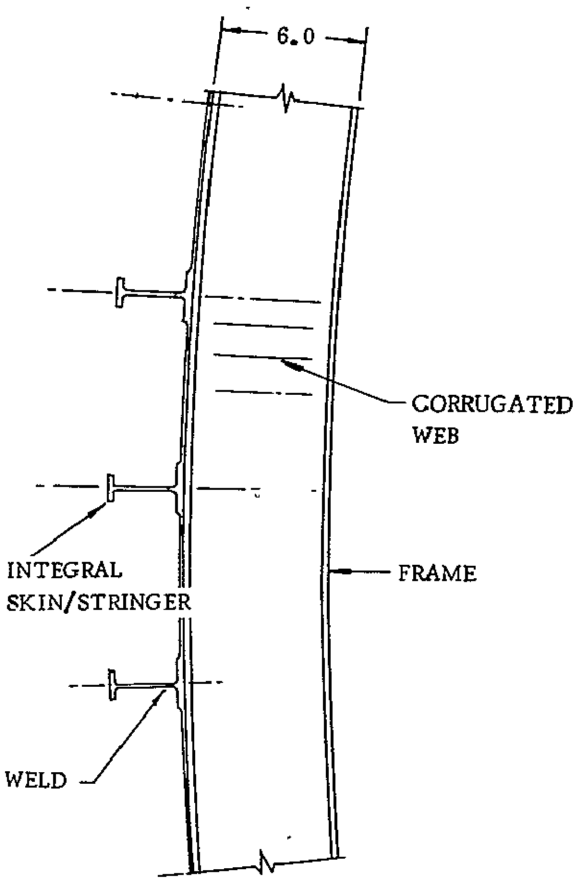


Figure 8-13. Thrust Skirt Detail

Figure 8-13 shows a diffusion-bonded skin assembly in which the integral stringers are short rectangular blades. These may be machined to a constant height in preparation for welding the T sections that comprise the major stringer areas. The T sections may be produced as extrusions and machined on all surfaces to produce the necessary finish and dimensional accuracy.

T sections provide a higher allowable stress level than "blade" stringer, and therefore have an inherent weight advantage. The skin-stringer weight of the T configuration is 20 percent less than the blade stringer configuration.

The thrust and holddown posts are subjected to high axial loads which are applied to the skin and immediately sheared circumferencially away from the areas of load concentration. Thus, skins and stringers immediately adjacent to the major structure attachments are of greater area or thickness.

"Lands" are incorporated in the panels for thrust ring attachments and for load redistribution around the five 21-inch-diameter cutouts that provide for propulsion lines.

An alternative skin/stringer configuration uses tapered machined titanium skin and is stiffened by alumimum-boron hat-section stringers. These stringers are prefabricated of unidirectional composite material bonded to the titanium skins. Both materials have a similar coefficient of expansion and will be limited to temperatures below 600°F.

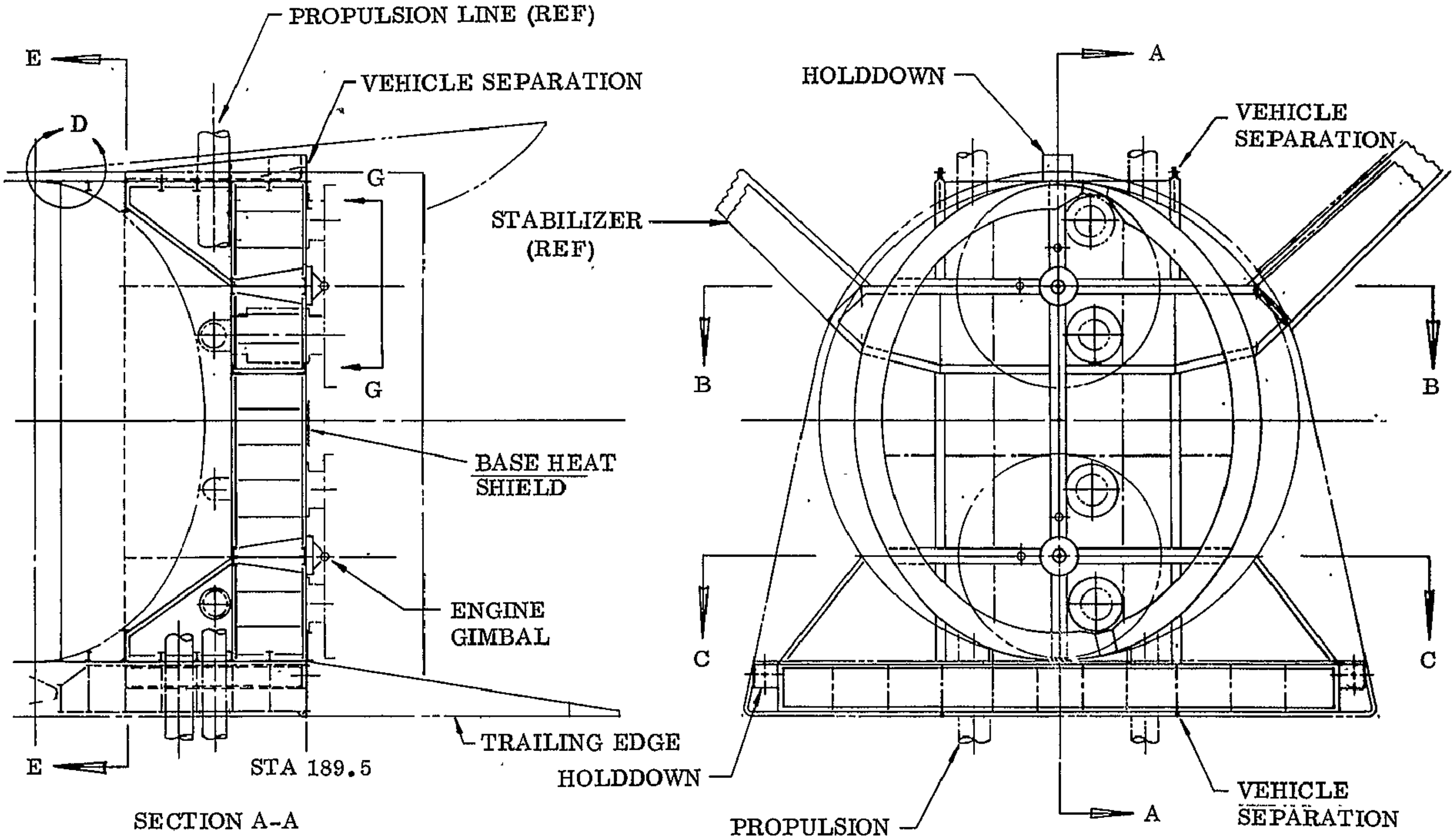

PROPULSION LINE (REF)
E
VEHICLE SEPARATION
D
G
G
BASE HEAT SHIELD
ENGINE GIMBAL
TRAILING EDGE
E
STA 189.5
SECTION A-A
A
HOLDDOWN
VEHICLE SEPARATION
STABILIZER (REF)
B
B
C
C
HOLDDOWN
A
VEHICLE SEPARATION
PROPULSION LINES (REF)

Figure 8-12. Aft Body Structural Arrangement

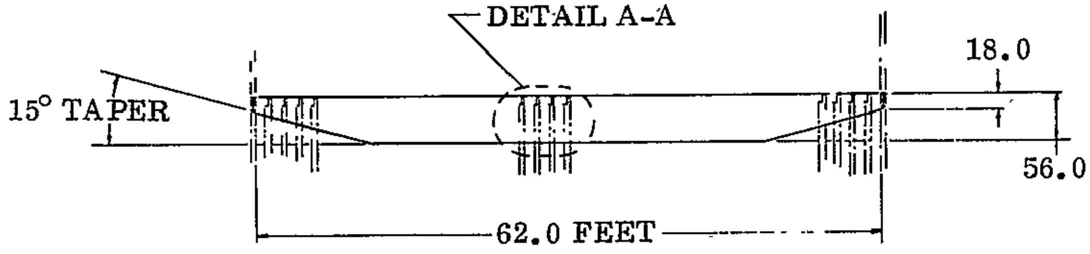
Payload door use criteria indicated that a door capable of supporting primary loads was not desirable. As a result, a longeron directly adjacent to the 17-foot-wide payload opening in the primary structure must support all loading conditions. The two longerons must replace in structural capability the nearly 21 feet of peripheral primary structure. A payload support structure extends from the longeron approximately five feet towards the base of the vehicle, and thus intersects the centerline of the payload envelope. It is intended that appropriately configured payload support fittings would be attached to this structure.

To uniformly distribute the loads, the longeron and payload support structure is tapered from a 5-foot maximum depth to an 18-inch depth at the landing gear bulkheads. The payload door longeron is backed up by a longeron that tapers from the landing gear bulkhead toward the integral structure propellant tanks. It is intended that both the door longerons and the tapered longerons in the tank adapter section can be removed for the booster. In addition, continuous frames will be spliced in for the booster, and the skin and stringers will be continuous in the area of the payload door opening.

8.4.9 AFT BODY STRUCTURE. The thrust structure is a cylindrical semi-monocoque shell with an overall diameter of 26.5 feet and a length of approximately 13.3 feet. The structural assembly is illustrated in Figure 8-12. The forward end of the skirt is bolted to the Y-shaped ring of the LH2 gank and provides primary structural continuity with the forward body of the vehicle.

The shell consists of a series of 20 integrally stiffened skin panels that use T-section stringers spaced at nominally 3.0 inches. Skin panels are welded together to form larger panels, which are installed between adjacent engine support beams. Seven inner frames, spaced at two-foot intervals, provide skin/stringer panel stability. Spacing is determined by the clearance required for the propulsion lines and structural continuity with the engine beam thrust structure. Fore and aft thrust rings within the thrust skirt are primary supporting members. They provide radial load restraint in the aft body structure to restrain the engines, the stabilizer supporting structure, holddown fittings, trailing edge support structure, vehicle separation linkage, and base heat shield.

The basic structure is fabricated from titanium alloy Ti 8-1-1, which is capable of withstanding temperatures of 600°F without serious structural degradation. Aluminum-boron composite material is used in areas where simple construction and considerable weight saving warrant. It is used with titanium structure for unidirectional members such as stringers and beam caps.


DETAIL A-A
18.0
15° TAPER
56.0
62.0 FEET

ORIENTATION

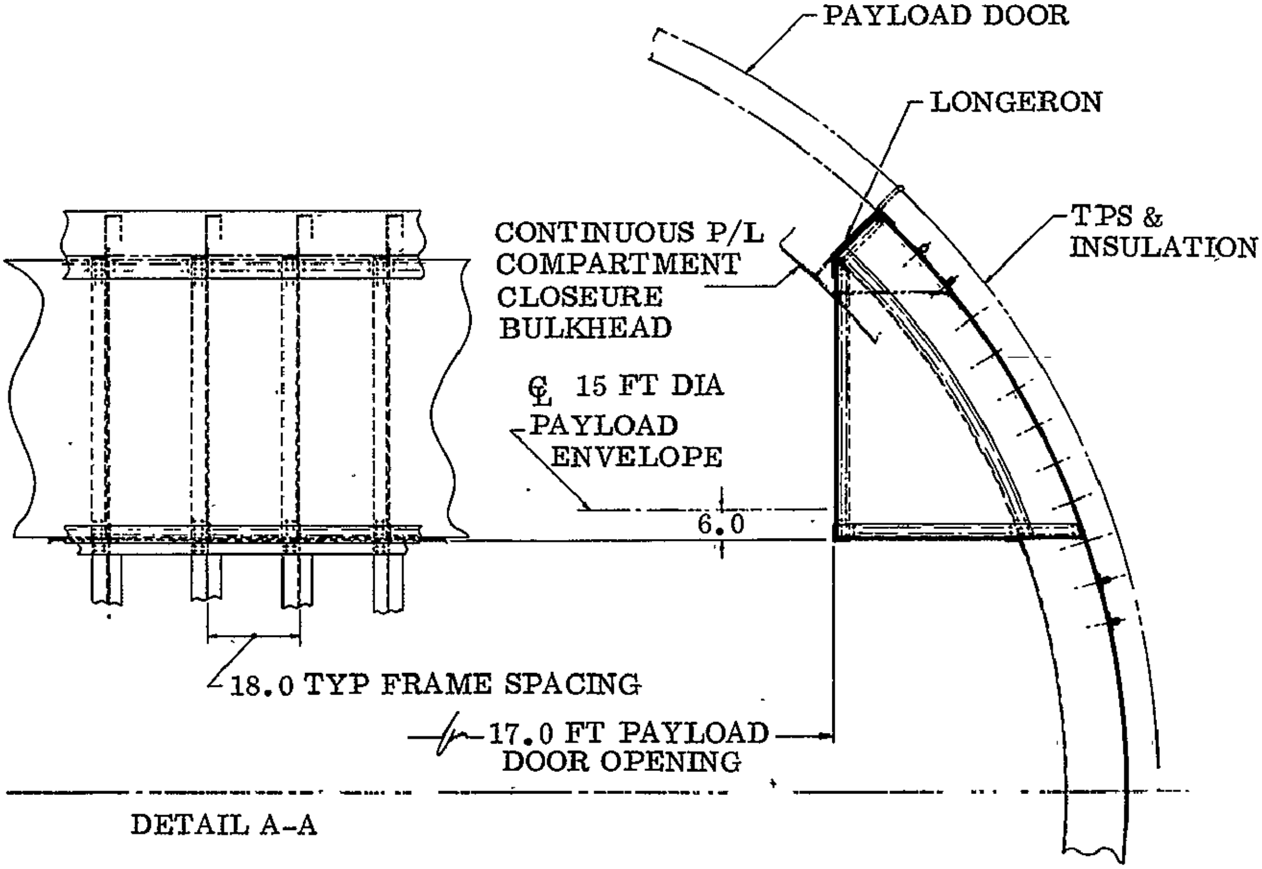

PAYLOAD DOOR
LONGERON
CONTINUOUS P/L COMPARTMENT CLOSEURE BULKHEAD
TPS & INSULATION
℄ 15 FT DIA PAYLOAD ENVELOPE
6.0
18.0 TYP FRAME SPACING
17.0 FT PAYLOAD DOOR OPENING
DETAIL A-A

Figure 8-11. Payload Door Longeron Arrangement

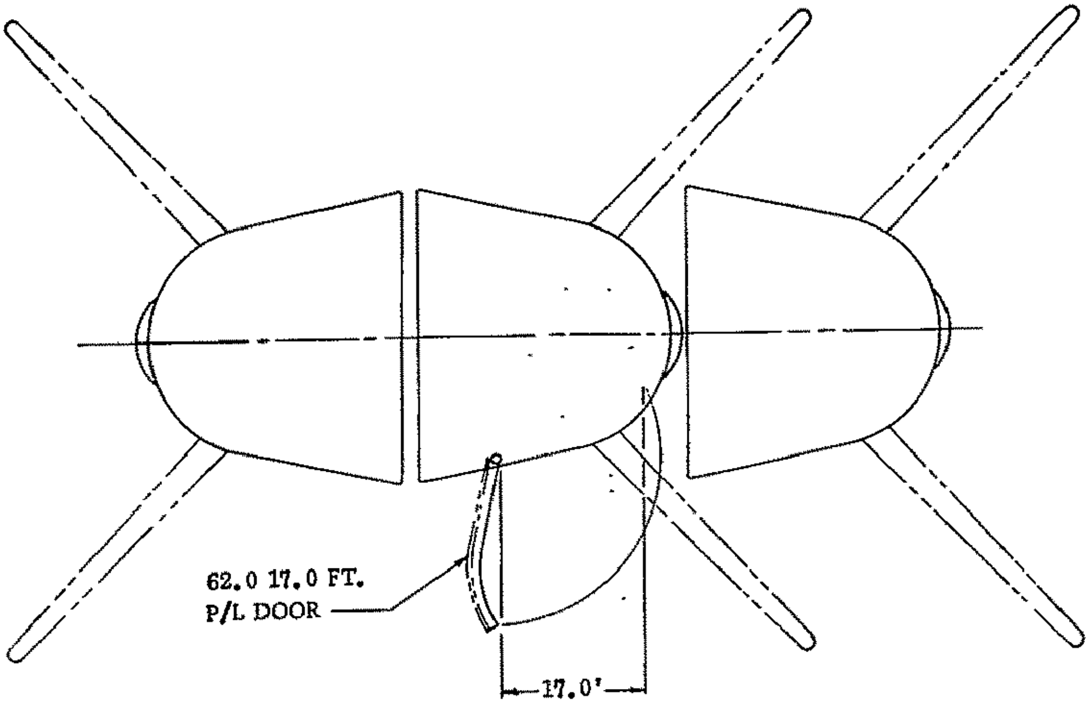

62.0 17.0 FT.
P/L DOOR
17.0'

Figure 8-9. **Side-Opening Payload Door Configuration**

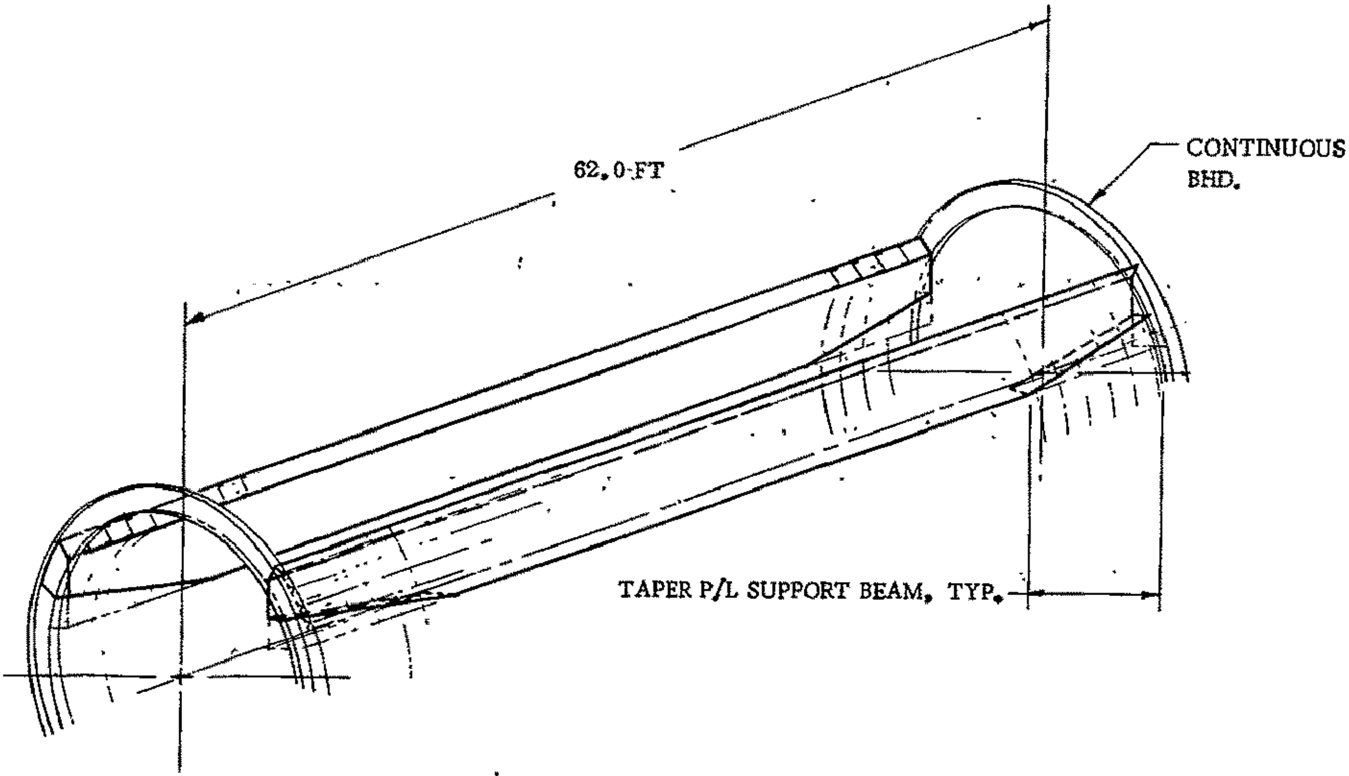
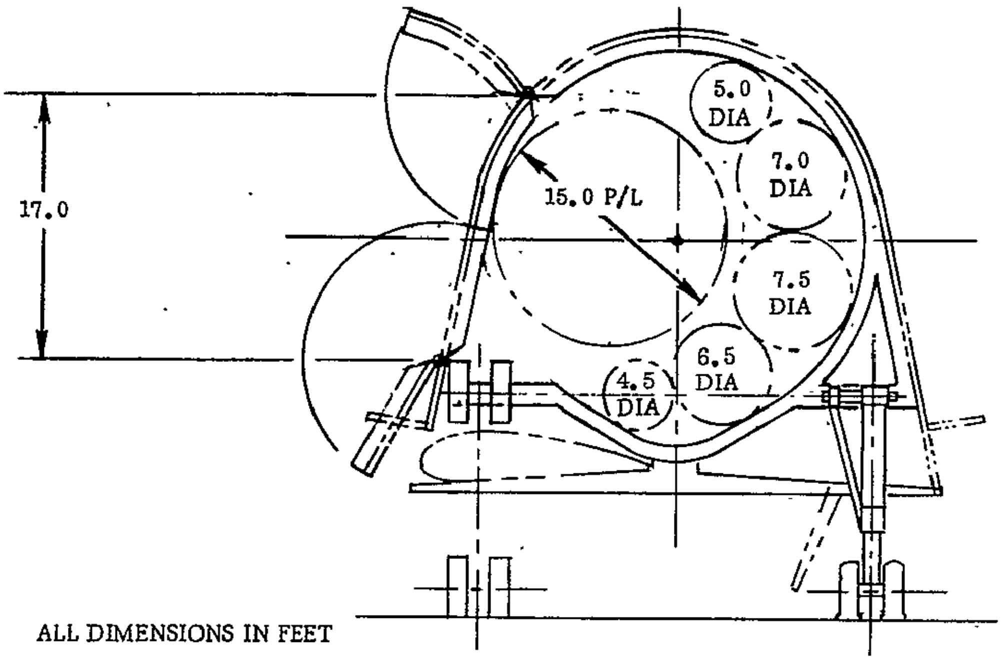
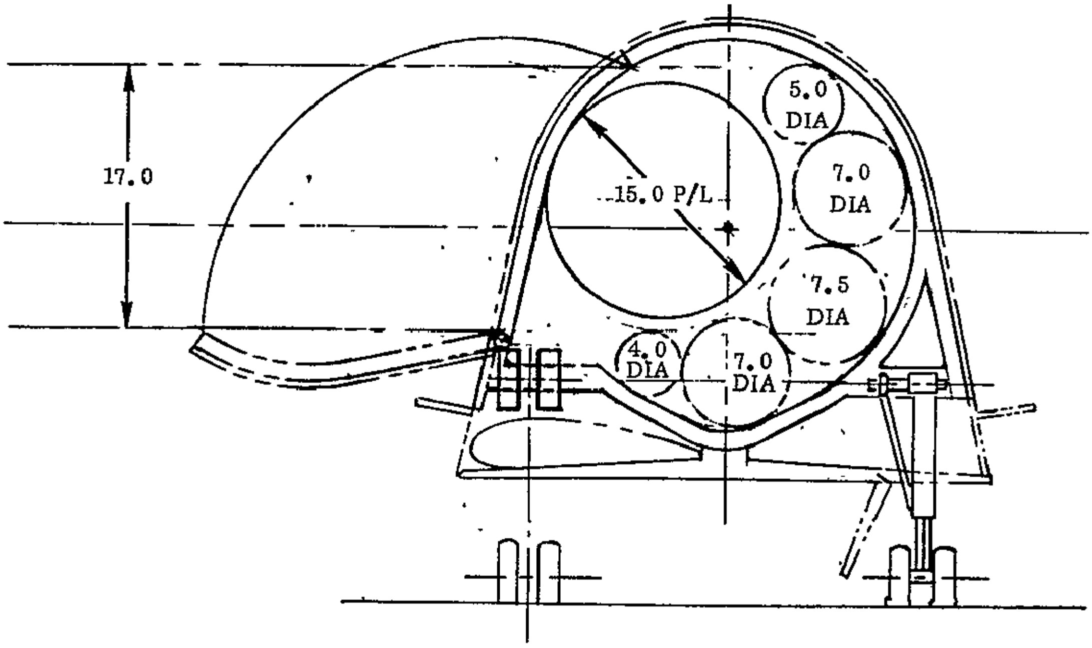

62.0 FT
CONTINUOUS
BHD.
TAPER P/L SUPPORT BEAM, TYP.

Figure 8-10. **Payload Door Longeron Geometry**


17.0
15.0 P/L
5.0 DIA
7.0 DIA
7.5 DIA
6.5 DIA
4.5 DIA
ALL DIMENSIONS IN FEET

Split-Door Concept


17.0
15.0 P/L
5.0 DIA
7.0 DIA
7.5 DIA
7.0 DIA
4.0 DIA

Single Door Concept

Figure 8-8. Side-Opening Payload Door Arrangement

slightly over 10 feet around the periphery and is 62 feet long and 6 inches thick. The inboard and outboard skins of the door are fabricated from two layers of titanium sheet, the inner door skins on each side are beaded for panel stability, and the skins are attached by spotwelding or diffusion bonding. A blanket of Microquartz insulation is adhesively bonded to the inboard surface of the door for thermal control in the payload compartment. The door is remotely controlled and activated hydraulically.

Latching at the interface of the doors requires that the door with threaded self-locating receptacles be closed. Next, the door with electrically operated threaded studs is closed. When the doors have been secured, an override switch turns off the threaded stud.

An investigation of the possibility of side-opening payload doors for the FR-1 configuration was conducted. (See Figures 8-8 and 8-9.)

One advantage of the side-mounted door is that the payload compartment is fully accessible during prelaunch operations after the complete Triamese launch vehicle has been erected. Disadvantages, however, include:

a. Large body bending loads from wing pivot and main landing gear are difficult to shear across door area unless door is made structural. Complexity of remotely controlled payload door is greatly increased by the requirement that it carry body structure loads (very complex hinge and latch mechanism with complexity proportional to opening angle required).

b. Vehicle lack of symmetry will cause control problems in launch, orbital, and cruise modes. The unsymmetric arrangement of centerbody propellant containers will result in a proportional increase in ACS requirements, and may cause a serious control problem.

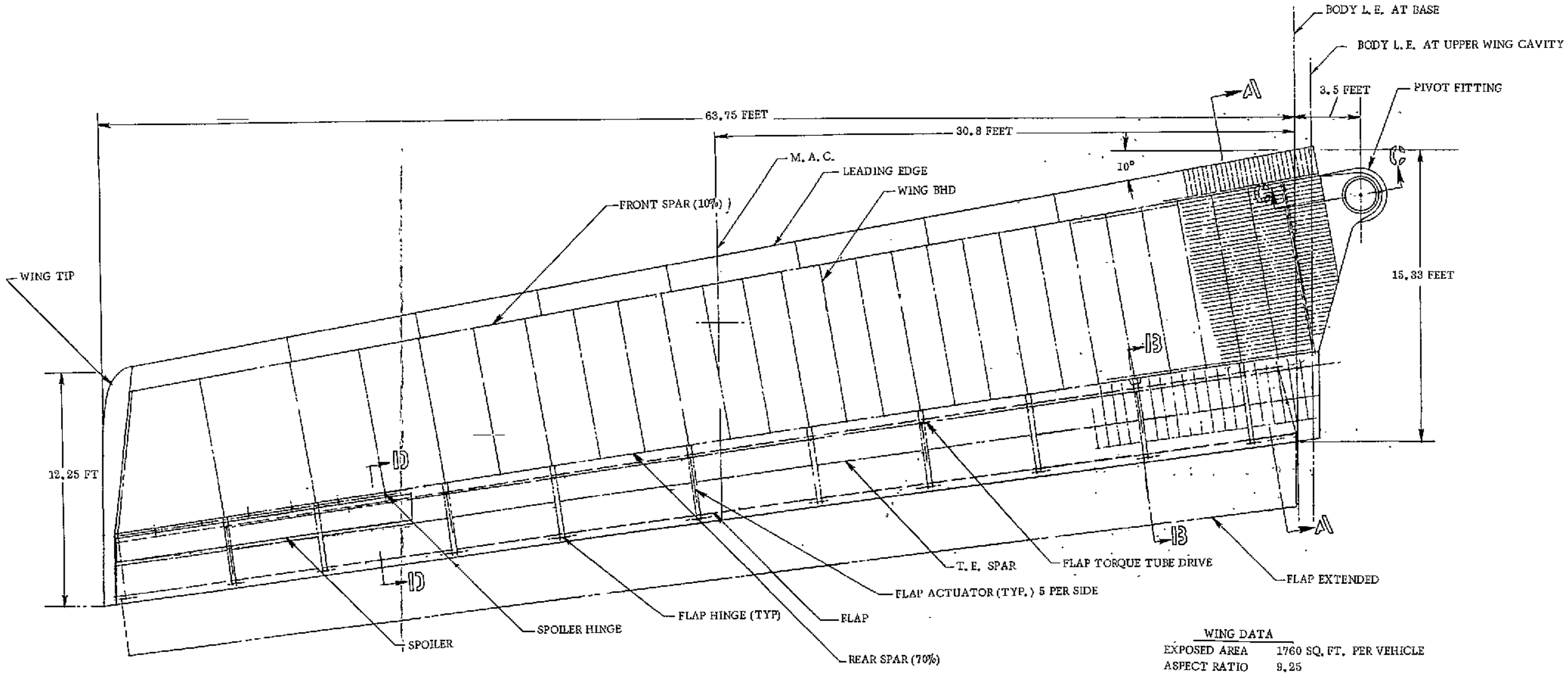
c. Expected higher heating rates along the sides (including so-called "streaking" effect) will cause additional sealing and latching problems.

d. The high temperatures may preclude the use of a "hot door structure", as is planned for a door located in the top of the vehicle. This would require the more difficult TPS-to-TPS sealed joint.

e. Unsymmetrical payload support structure would be provided at a great loss in structural efficiency.

It was concluded (at least from the structural standpoint) that a side-opening payload door would be very undesirable, and a baseline top-opening bay was selected.

The payload door longeron and integrally attached payload support structure (Figures 8-10 and 8-11) extends the 62-foot length of the payload door opening. Although further tradeoff studies are required to select the optimum material and configuration for the longeron, the current investigation indicates that this may be an ideal application for composite materials


BODY L.E. AT BASE
BODY L.E. AT UPPER WING CAVITY
A
3.5 FEET
PIVOT FITTING
63.75 FEET
30.8 FEET
℄
10°
M.A.C.
LEADING EDGE
WING BHD
FRONT SPAR (10%)
WING TIP
15.33 FEET
B
12.25 FT
D
D
B
A
T.E. SPAR
FLAP TORQUE TUBE DRIVE
FLAP EXTENDED
FLAP ACTUATOR (TYP.) 5 PER SIDE
FLAP HINGE (TYP)
FLAP
SPOILER HINGE
SPOILER
REAR SPAR (70%)

WING DATA	
EXPOSED AREA	1760 SQ. FT. PER VEHICLE
ASPECT RATIO	9.25
λ	0.80
M.A.C.	13.9 FEET
ROOT AIRFOIL	NACA 4421
TIP AIRFOIL	NACA 4418

Figure 8-21. Wing Plan View, General Arrangement

On the base aft of the wings on both the boosters and orbiters, the frames of the aft body structure extend down to the TPS pressure barrier. The pressure barrier and sub-beams, but no main TPS support beams, need to be added.

Forward of the LO_2 tank, on all of the curved top over the tanks, cargo, and thrust structure, and on the sides aft of the LH_2 tank, the TPS is mounted on fittings attached to the basic structure of the orbiters and boosters. Posts or adapter fittings function both as structural supports and thermal standoffs.

The heat shields on the sides and top of the orbiters and on the base of the boosters have a much higher strength at their maximum operating temperatures than do the heat shields on the bases of the orbiters. Therefore, these lower temperature heat shields can accommodate the required Δp's without the necessity of an additional pressure barrier on the cold side of the insulation. The greater strength of these heat shields at operating temperatures also eliminates the need for sub-beams and additional supports between the main support frames.

The aft closure bulkhead provides the support for the TPS that protects the vehicle from the rocket engine heat.

8.4.11 WING STRUCTURE. The wing is a deployable type which pivots and is actuated at the inboard end. When stowed (retracted), it is fully enclosed in the vehicle body. The design and construction are accomplished by using existing technologies. Figure 8-21 shows the general arrangement of the wing.

Figure 8-22, which is Section A-A of Figure 8-21, shows a typical structural cross-section. The structural box consists of two spars, with the front spar located at 10 percent and the rear spar at 70 percent of wing chord. The spars are web type with T-type caps and J-type stiffeners. The plating is integral stiffened sheet with T-type stringer stiffeners on the upper surface and blade-type stiffeners on the lower surface. The bulkheads are the truss type with cruciform-section truss diagonal members and Z-type rails. Stringer clips attach every other stringer to the bulkheads. T-type intercostals are located in the corner areas of the bulkheads. Ti 8-1-1 material was selected for the wing box because it has the highest strength-to-weight ratio for both tension and compression members, and because it can be readily fusion-welded with inert gas protection.

The flaps are the "inverted" or "reversible" type (Figure 8-23). A fixed hinge is located near the wing trailing edge and is an integral part of the flap support bulkhead. The flap is supported at the hinge and two actuating links, with the upper line pivot supported by the flap support bulkhead. The nut of the actuator screw is trunnion-mounted in the upper link, and the flap is actuated as the nut travels along the revolving screw. The actuators are driven by a torque tube from a central power source. Five actuators are required per wing per side. The flap is built in segments, with

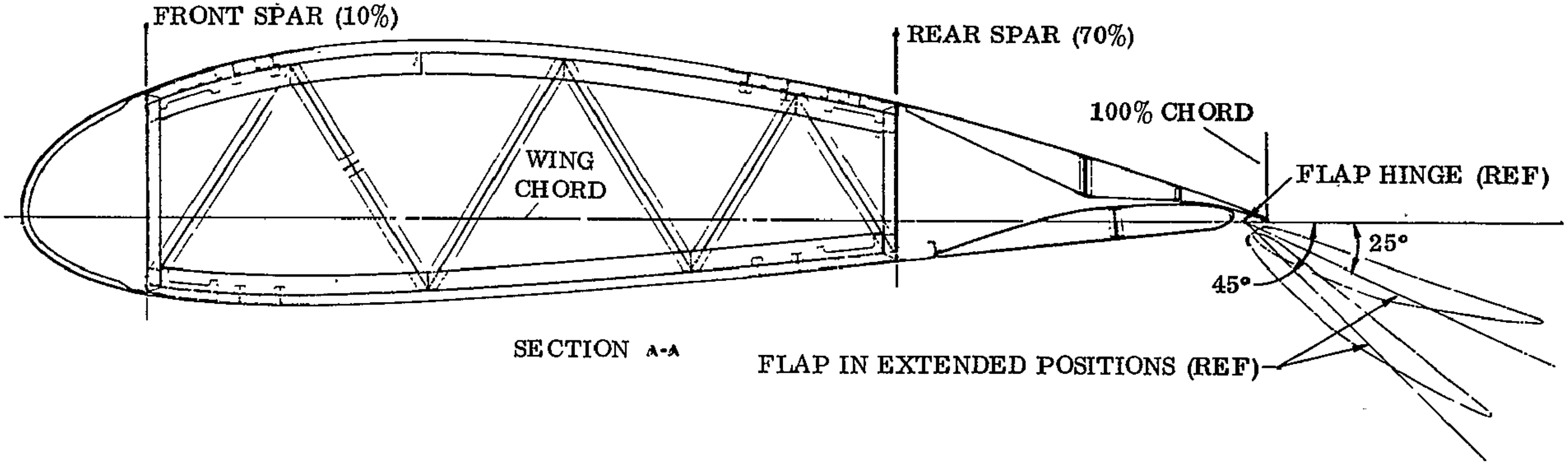

FRONT SPAR (10%)
REAR SPAR (70%)
100% CHORD
FLAP HINGE (REF)
WING CHORD
25°
45°
SECTION A-A
FLAP IN EXTENDED POSITIONS (REF)

Figure 8-22. Wing Cross-section

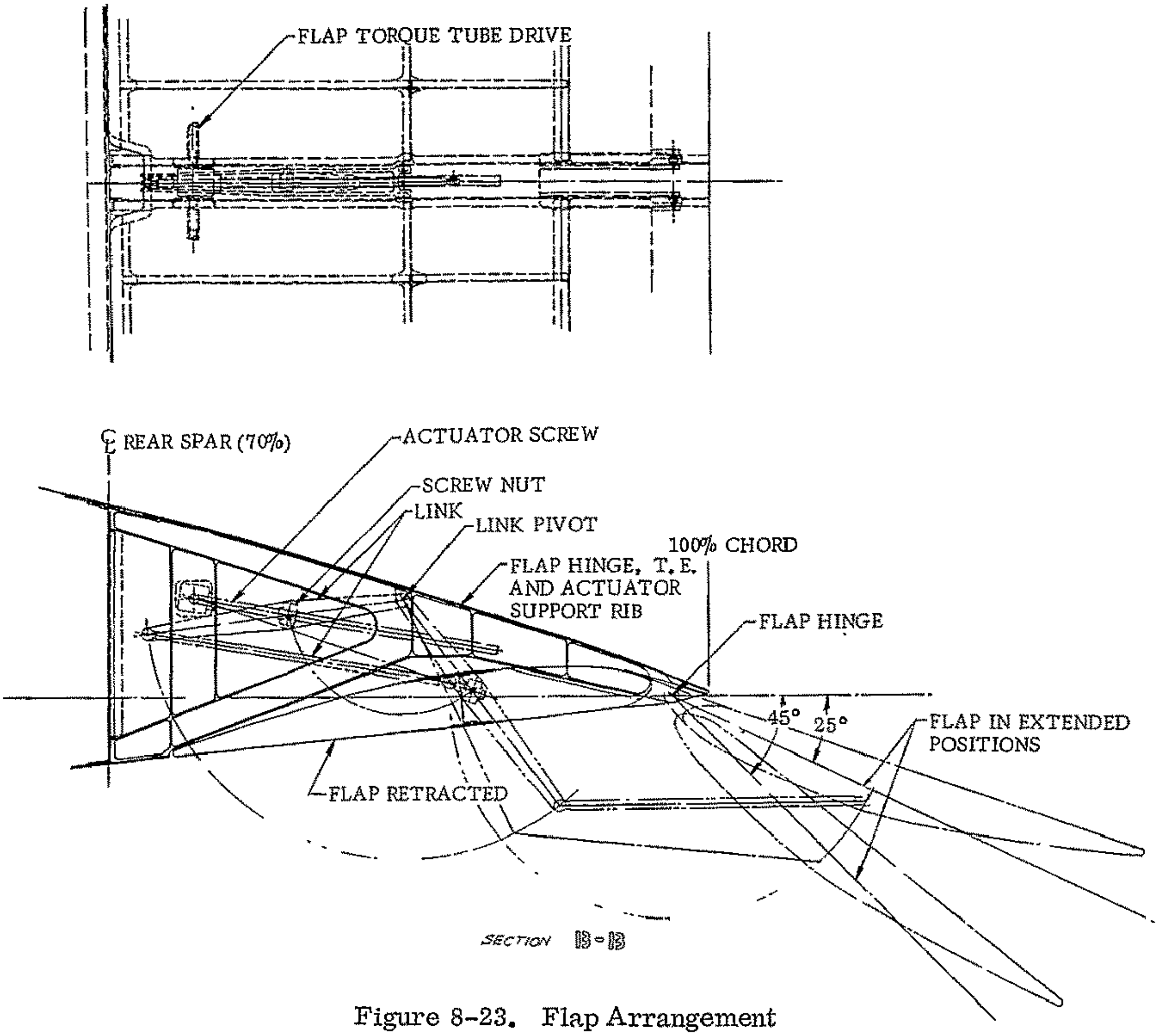
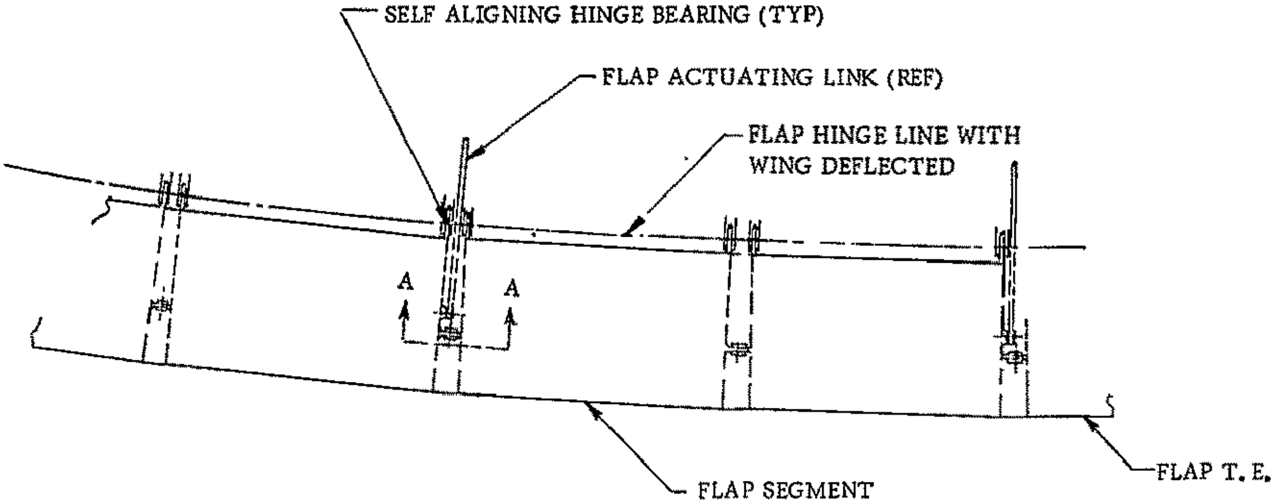

FLAP TORQUE TUBE DRIVE
℄ REAR SPAR (70%)
ACTUATOR SCREW
SCREW NUT
LINK
LINK PIVOT
100% CHORD
FLAP HINGE, T.E. AND ACTUATOR SUPPORT RIB
FLAP HINGE
45°
25°
FLAP IN EXTENDED POSITIONS
FLAP RETRACTED
SECTION B-B

Figure 8-23. Flap Arrangement

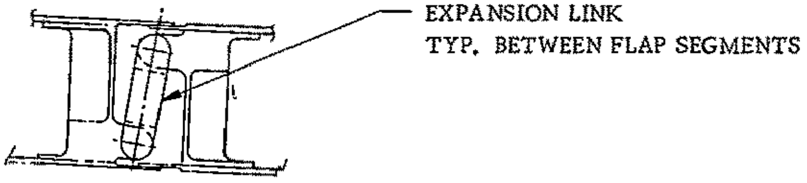
expansion links between segments and self-aligning hinge bearings to accommodate wing deflections. (See Figure 8-24.) This type of flap was selected because of its efficiency, simplicity, and minimum weight penalty.

The flap is constructed of a single spar and ribs with sheet skins. Each segment is about 70 inches long, with machined end ribs that include hinges and expansion link and actuator link support provisions. Aluminum alloy was selected because of the minimum gages required for the flap structure and the maximum temperature environment of 200°F.

The wing trailing edge consists of upper and lower surface panels. (See Figure 8-22.) The upper surface panel has a spar near the center, with ribs forward and aft. A trailing-edge wedge of honeycomb construction is supported by the aft ribs and skin. The trailing-edge upper surface is made in segments supported at their ends by the flap support bulkheads. The forward edge is supported by the wing rear spar. This design provides for maximum access in the trailing edge and fabrication and assembly


SELF ALIGNING HINGE BEARING (TYP)
FLAP ACTUATING LINK (REF)
FLAP HINGE LINE WITH WING DEFLECTED
A
A
FLAP SEGMENT
FLAP T. E.

VIEW LOOKING FWD
FLAPS EXTENDED, WING DEFLECTED UP


EXPANSION LINK
TYP. BETWEEN FLAP SEGMENTS

Section A-A

Figure 8-24. Flap Expansion Joint

efficiencies. The flap support bulkheads (Figure 8-23) are machined and include integral flap hinges and provisions for supporting the trailing edge and flap-actuator link attachment.

The lower-surface trailing edge is located between the wing rear spar and the flap. It consists of skin and stiffeners and a channel beam along the aft edge supported at the ends by the flap support bulkheads. Aluminum alloy was selected for the trailing edges because of minimum gage requirements and the maximum temperature environment of 200°F.

The spoiler is located just aft of the wing rear spar. It is 10-percent of the wing chord and 25 percent of the outer span. It is constructed of aluminum alloy honeycomb and aluminum alloy hinge fittings, and is hydraulically actuated (Figure 8-25). Aluminum alloy was again selected because of its strength-to-weight ratio and because of the maximum environmental temperature of 200°F.

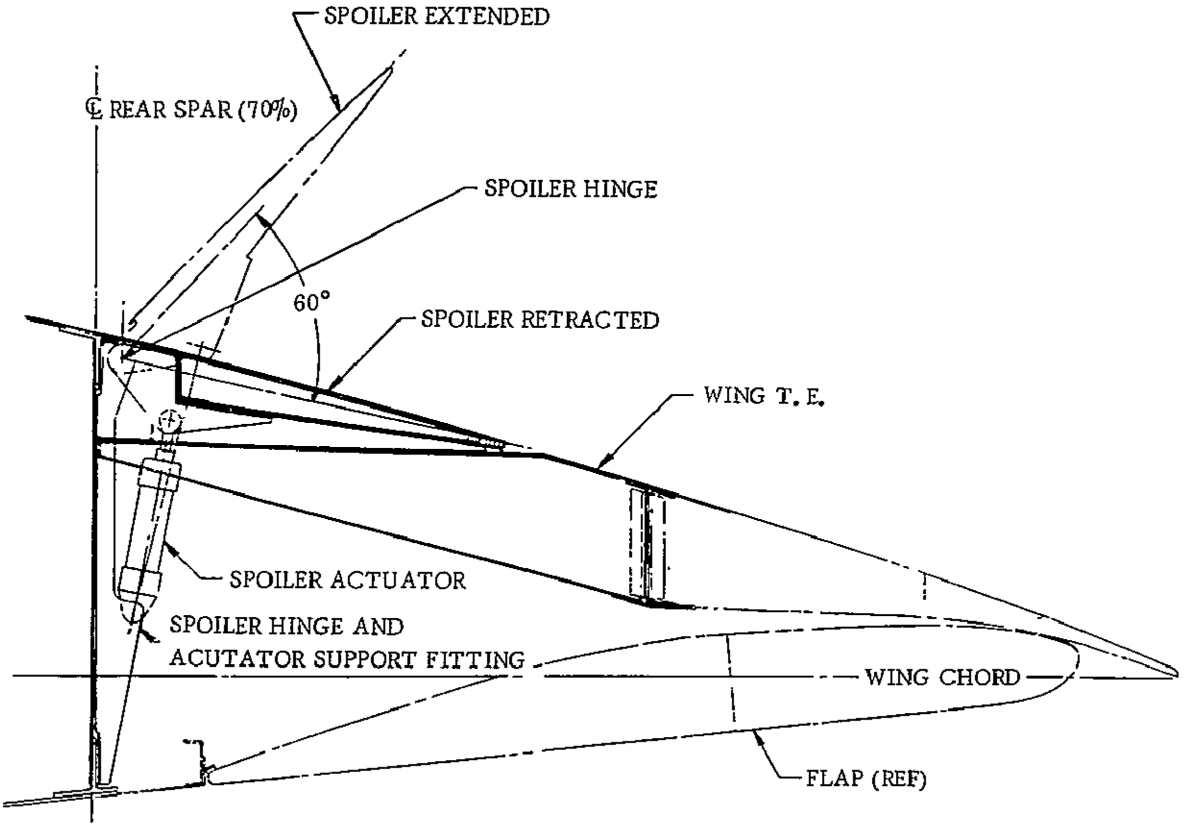


Figure 8-25. Spoiler Arrangement

The wing pivot fitting is attached to the inboard end of the wing box (Figure 8-26). The fitting is attached to the wing box within a span of 24 inches. Improvements on F-111 wing pivot technology include lower bearing stresses and the double shear feature on the pivot pin. The fitting is built-up of machined titanium parts welded together. The upper and lower inboard lugs provide the housing for large self-aligning bearings.

A shear-takeout fitting is located outboard and between the pivot fitting lugs. It houses a self-aligning shear receptacle fitting. A slot in this shear receptacle fitting accommodates a shear lug tongue. The shear lug is attached to the body bulkhead. As the wing pivots, the shear receptacle fitting slides on the shear lug tongue and the wing shear is transferred to the body bulkhead; no vertical shear is carried in the pivot lugs. Ti 8-1-1-1 material was selected for the wing pivot fitting because it has the highest strength-to-weight ratio for both tension and compression and because it can be readily fusion-welded with inert gas protection.

The wing leading edge is 10 percent of the wing chord. It consists of an outer skin and a corrugated inner skin, with ribs approximately every 20 inches, and with extruded attach strips. Aluminum alloy was selected because of minimum gage requirements and the maximum environmental temperature of 200°F.

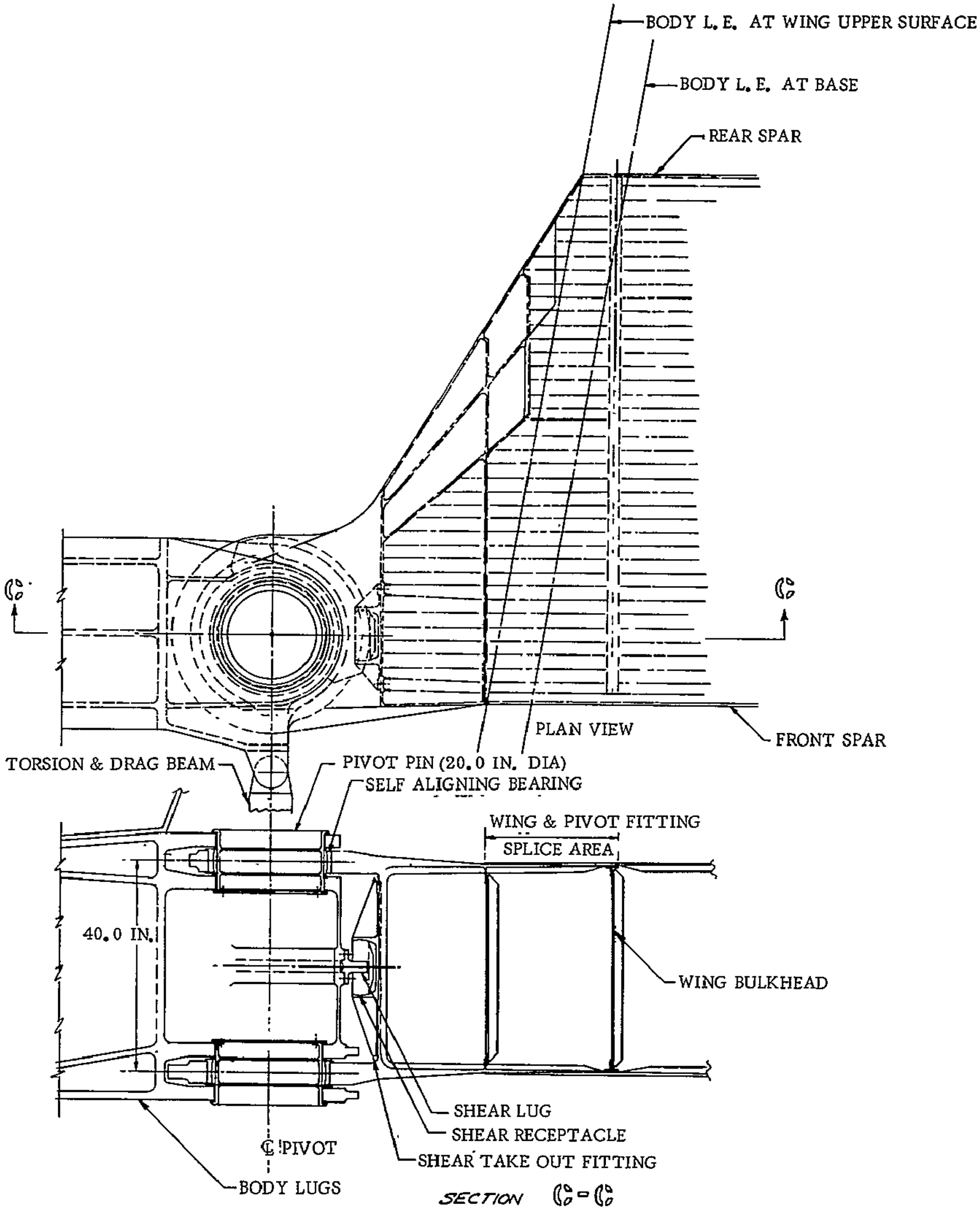

BODY L. E. AT WING UPPER SURFACE
BODY L. E. AT BASE
REAR SPAR
C
C
PLAN VIEW
FRONT SPAR
TORSION & DRAG BEAM
PIVOT PIN (20.0 IN. DIA)
SELF ALIGNING BEARING
WING & PIVOT FITTING
SPLICE AREA
40.0 IN.
WING BULKHEAD
SHEAR LUG
SHEAR RECEPTACLE
SHEAR TAKE OUT FITTING
℄ PIVOT
BODY LUGS
SECTION C-C

Figure 8-26. Wing Pivot Fitting Arrangement

8.4.12 STABILIZER STRUCTURE

Orbiter

There are two fixed stabilizers mounted on the vehicle body to form a V-type tail. The structure is fully protected thermally except for the leading edge nose cap, which is coated tantalum. The structure consists of a rudder, structural box, and a leading edge. (See Figures 8-27 and 8-28.)

The rudder is full span, and is 35 percent of the stabilizer chord. It consists of front and rear spars with corrugated (sinusoidal) webs and channel-type caps. The ribs are of similar construction. The upper and lower skins are similar; each consists of an outer and inner skin, with the inner skin corrugated. A trailing-edge wedge skin is similar in construction to the upper and lower skins. The rudder nose consists of a nose skin and ribs with cutouts at the hinges. The hinge fittings carry self-aligning bearings. The rudder consists of two segments and is actuated at three locations along its span. Material is Ti 8-1-1 except for the hinge fittings, which are steel.

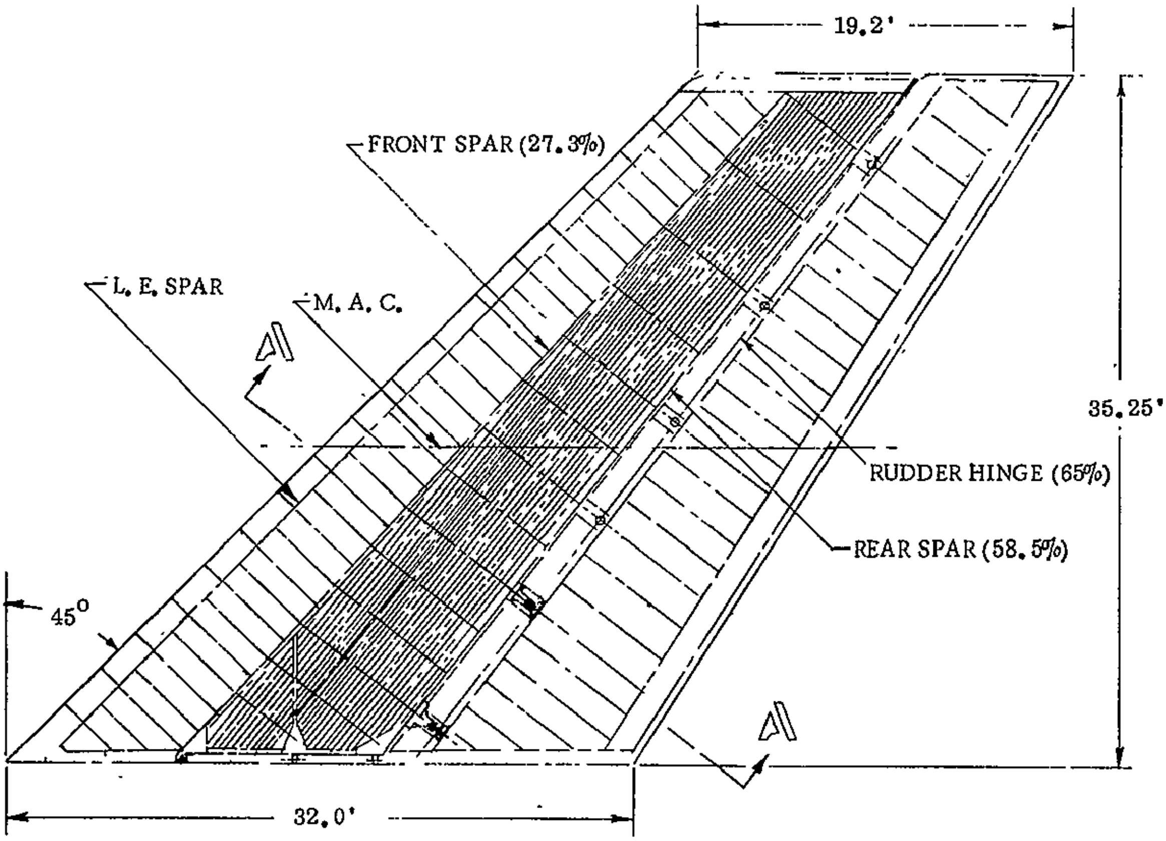
The structural box is located between 27.3 and 58.5 percent of the stabilizer chord. The upper and lower plating consists of integral-stiffened titanium sheet with Z-type stringer stiffeners. The bulkheads consist of corrugated (sinusoidal) webs and channel-type caps. All box material is titanium except the rear and intermediate spar attach fittings, which are steel.

The rudder nose shroud is attached to the structural box at the rear spar. The skins consist of an outer skin and a corrugated inner skin and ribs. Material is Ti 8-1-1, and steel hinge support fittings attach to the box skins and rear spar.

The leading edge between the nose cap and the box front spar consists of ribs and spars with corrugated (sinusoidal) webs and channel-type caps, and skins consisting of an outer skin and a corrugated inner skin. All material is titanium. Sinusoidal spar and rib webs are used to provide relief for the thermal stresses.

Titanium 8-1-1 was selected because of its strength-to-weight ratio under the environmental temperatures and because it can readily be fusion-welded with inert gas protection and spot-welded without atmospheric protection with weld strength comparable to that of the parent material.

The leading edge nose cap consists of an outer skin and a corrugated inner skin and ribs. The nose cap is made in segments with provisions for thermal expansion. The material is coated tantalum.


19.2'
FRONT SPAR (27.3%)
L. E. SPAR
M. A. C.
A
35.25'
RUDDER HINGE (65%)
REAR SPAR (58.5%)
45°
A
32.0'

STABILIZER DATA

EXPOSED "V" TAIL

AREA	904.5 SQ. FT.
SPAN	35.25 FT
ASPECT RATIO	1.375
AIRFOIL ROOT	NACA 4412 MOD.
TIP	NACA 4410 MOD.
L. E. RADIUS	6.00" CONSTANT

Figure 8-27. Stabilizer Plan View, Structural General Arrangement

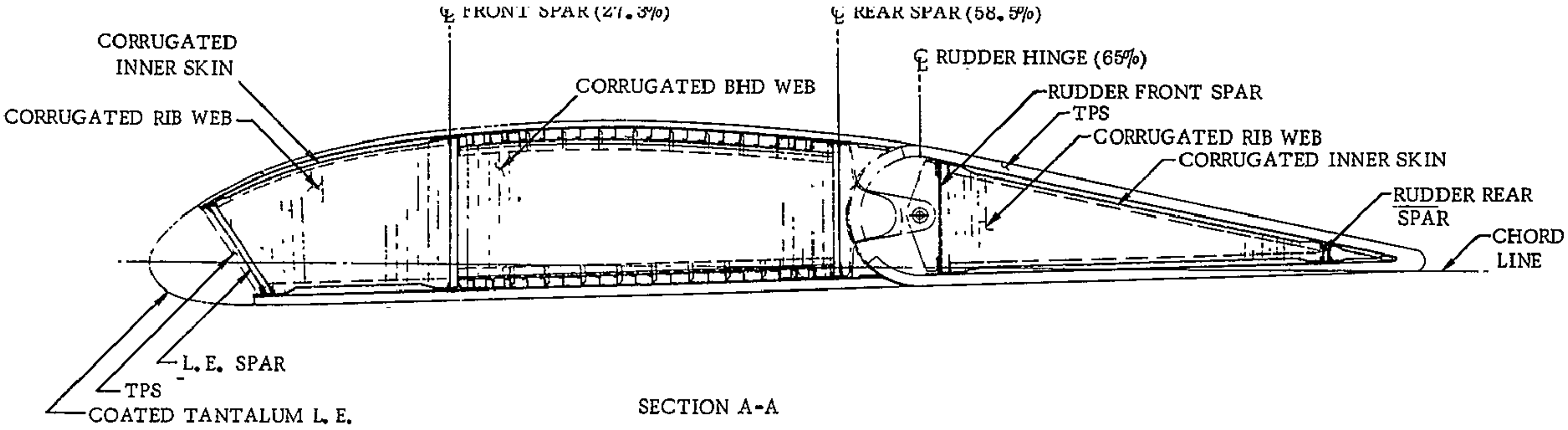

℄ FRONT SPAR (27.3%)
℄ REAR SPAR (58.5%)
℄ RUDDER HINGE (65%)
CORRUGATED INNER SKIN
CORRUGATED RIB WEB
CORRUGATED BHD WEB
RUDDER FRONT SPAR
TPS
CORRUGATED RIB WEB
CORRUGATED INNER SKIN
RUDDER REAR SPAR
CHORD LINE
L. E. SPAR
TPS
COATED TANTALUM L. E.
SECTION A-A

Figure 8-28. Fin Cross Section

The thermal protection used aft of the nose cap on the lower surface (including both surfaces of the rudder) will be one-third dynaflex on the outer surface and two-thirds microquartz on the inner surface of a 2.00-inch-thick layer. An L605 heat shield will be used as the cover. The upper surface thermal protection will be similar except that it is only 1.00 inch thick.

Booster

There are two fixed stabilizers mounted on the vehicle body to form a V-type tail. The structure is of the "hot" type. The structure consists of a rudder, structural box, and a leading edge.

The construction and arrangement is similar to the orbiter stabilizer except it has no thermal protection. The material throughout is 718 alloy.

8.4.13 INTERSTAGE CONNECTIONS. The stage interconnect consists of attachment points at three places between each booster and the orbiter. One is located at Station 72 on the Y-direction centerline and the other two are located at Station 190, 13 feet apart.

The forward attachment point at Station 72 is located at the forward bulkhead of the payload bay of the orbiter vehicle (Figure 8-29). The interface at this point is in the plane of the station and is inside the outer contour of the orbiter vehicle. The flanged fittings are held together by studs and explosive nuts. After separation, no protrusions would exist on the orbiter; only an indentation in the surface at the centerline on both the top and bottom of the vehicle. Structure extends from the booster across the two-foot space, and a hinged thermal-protection fairing covers this structure after vehicle separation for the return flight.

The forward attachment point carries interconnect loads in three directions: axial (X), lateral (Y), and vertical (Z). The largest load (1,480,000 pounds ultimate), occurring at booster burnout is the net thrust transferred from the booster to the orbiter. This X-direction force is reacted in compression at the interface, and the Z-direction force resulting from the moment is reacted by shear in the studs. Most of the axial load into the orbiter is transferred to the LO_2 tank. A longeron carries the load from the interface fitting and shears it into a thick aluminum skirt (0.30 to 0.125 inch) around the base of the LO_2 tank. Frames inside the skirt stabilize the skirt and react the moments created as the compressive load is fed into the LO_2 tank bulkhead. The forward payload bay bulkhead or frame reacts the Z- and Y-direction loads. The net thrust from the booster is transferred to the interface fitting by a thrust column. A frame is added to the booster at Station 89.5 to react resultant inward force. A longeron is added along the bottom center line of the shell on the booster to shear load from the shell. The outward resultant force from the thrust column and the Z- and Y-direction loads are reacted into the frame at Station 72.

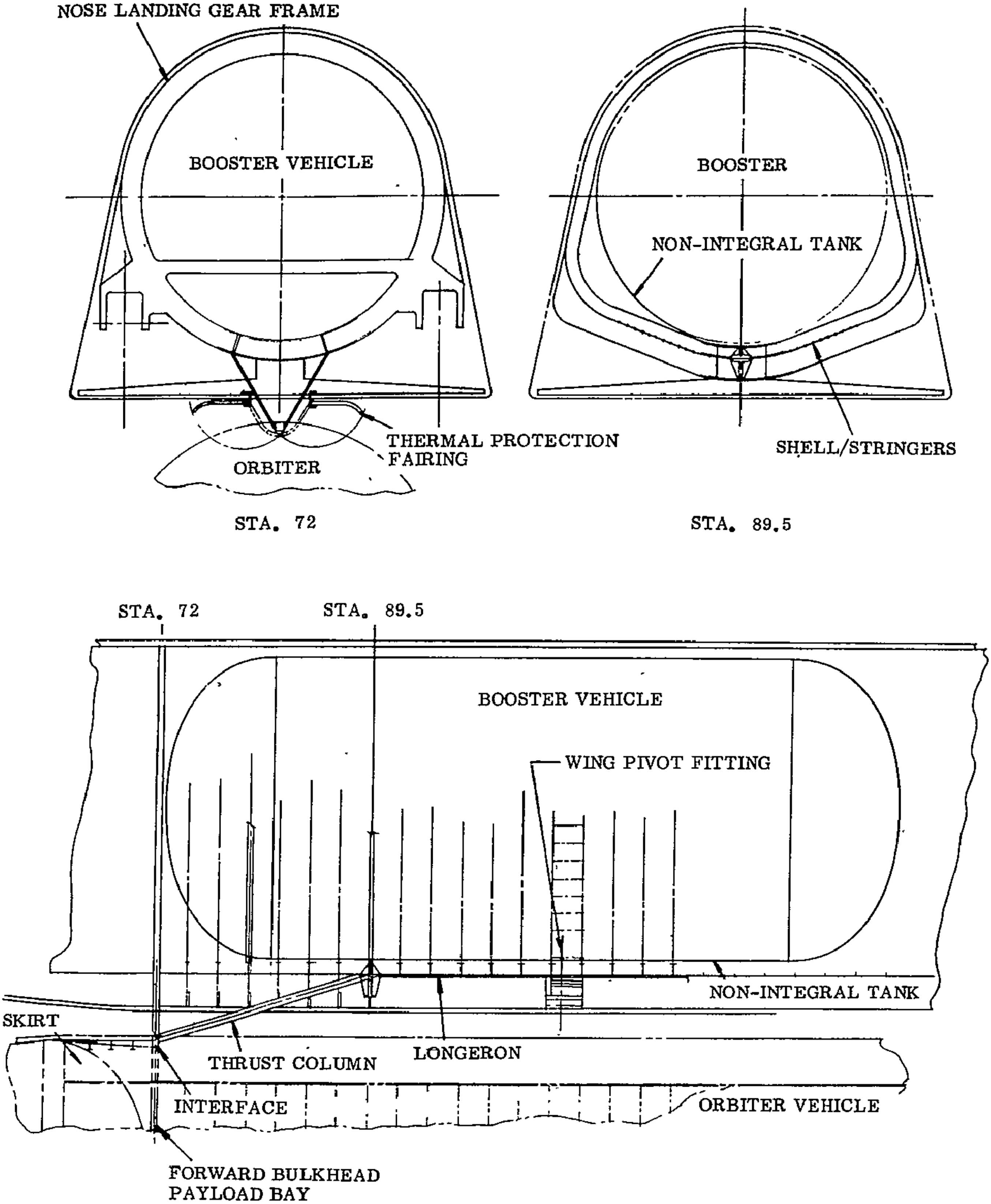

NOSE LANDING GEAR FRAME
BOOSTER VEHICLE
THERMAL PROTECTION FAIRING
ORBITER
STA. 72
BOOSTER
NON-INTEGRAL TANK
SHELL/STRINGERS
STA. 89.5
STA. 72
STA. 89.5
BOOSTER VEHICLE
WING PIVOT FITTING
NON-INTEGRAL TANK
SKIRT
THRUST COLUMN
LONGERON
INTERFACE
ORBITER VEHICLE
FORWARD BULKHEAD
PAYLOAD BAY

Figure 8-29. Forward Attachment Point Arrangement

The aft attachment points are located at the aft frame of the thrust structure at Station 190. (See Figure 8-30.) They carry tension loads between the vehicles that are the resultant forces from the moment created by transferring thrust from the booster to the orbiter, and they carry lateral loads between the vehicles. These points also form the hinge line for the booster to rotate away from the orbiter during the separation sequence. The two aft points have their interface at the outer contour line of the orbiter. Two tension members and the stage separation linkage extend from the booster to the attachment points on the orbiter.

Before the staging sequence is initiated, the two tension members carry Z-direction forces and the stage separation linkage carries lateral (Y-direction) forces. At initiation of staging, the two tension links are separated simultaneously with the forward attachment point. The stage separation linkage is still attached.

The stage separation linkage consists of rotating links, folded links, and a displacement-control member. The function of the linkage is to prevent interference between vehicles as the booster rotates about the pivot points during the stage separation sequence. The booster inertia and the aerodynamic forces cause the booster to rotate. The linkage unfolds during the rotation phase of separation, providing clearance between the aft ends of the vehicles. The displacement-control member controls the rate at which the folded link unfolds to extend the linkage. The forces in the linkage are such that the displacement-control member (which is a hydraulic snubber) is always in tension during the rotation phase.

The linkage, including pivot fittings, are separated at the interface line when the booster has reached the desired rotation angle. When separation is complete, the linkage system is retracted by the displacement-control member into the booster vehicle heat shield. The members of the linkage have thermal protection on one side so that when retracted the booster lower surface is flush and thermal protected. The linkage is isolated from the aft end structure by a thin-gage heat-resistant metal box to prevent hot gases from penetrating the vehicle if the seals around the retracted linkage leak.

The linkage is attached to the booster vehicle at five points. Three are located at Station 177.5, where the centerline point is for the displacement control member and the two side points are for the rotating links. Three beams at these locations run the length of the thrust skirt to shear the axial direction load into the thrust skirt. The resulting moments are reacted in the forward and aft frames of the thrust structure. The Z-direction and lateral loads are also reacted by these frames. The two remaining points are located 6.5 feet from each side of the centerline on Station 190. Two of the longitudinal beams extend aft to pick up the pivot fittings for these two points. The Z-direction and lateral loads are carried into the aft thrust frame at Station 189.5.

8.4.14 THERMAL PROTECTION SYSTEM. The thermal protection system (TPS) surrounds the basic structure of the vehicle and protects it from the heat of reentry. The aft end of the vehicle requires thermal protection from the heat of the rocket engines. The outer surface of the TPS establishes the aerodynamic shape of the vehicle.

The TPS consists generally of insulation, heat shields, posts, and a supporting structure. The insulation performs the primary function of protecting the basic structure and vehicle contents from excessive temperature. The heat shields establish the aerodynamic contours of the vehicle and protect the insulation from the windstream and water. The posts mount on the supporting structure and protrude through the insulation to support the heat shields. They also provide support for the insulation. The supporting structure extends from the basic vehicle structure to the TPS cold surface. It is discussed separately in Section 8.4.10.

The fibrous insulation is Johns-Manville 4.5 pcf Microquartz and 8.0 pcf Dynaflex. Insulation thickness varies on the local heat shield temperature. The required thickness is generally built up with multiple layers, using Microquartz for the insulation layers where temperature will never exceed 1600°F. Dynaflex is used for the outside layers of insulation, where the temperature during service may exceed 1600°F.

The material used in the heat shields depends on the maximum reentry temperature. Heat shield temperatures vary with the reentry trajectory and with the emissivity of the heat shield surface; temperature requirements are defined in Section 8.2. Maximum design temperatures and emissivities of the materials are:

Coated T-222 tantalum, 2500 to 3100°F, $\epsilon = 0.80$

Coated C-129Y columbium, 2200 to 2500°F, $\epsilon = 0.80$

TD NiCr, 2000 to 2200°F, $\epsilon = 0.85$

Hastelloy X, 800°F to 1500°F, $\epsilon = 0.85$

HS-188 cobalt base alloy, 1500 to 2000°F, $\epsilon = 0.85$

6 Al-4V titanium alloy, 800°F and below, $\epsilon = 0.85$

The materials generally have short-time capabilities above the temperatures listed, but these are considered good design ranges; emissivities are also conservative. The heat shield temperature varies inversely as the fourth root of the emissivity.

The orbiters for the FR-3 and FR-4 have the same temperature and TPS requirements. The boosters are of somewhat different configurations. However, the heat shield temperatures differ so little that the heat shield materials are generally the same for the FR-3 and FR-4 boosters.

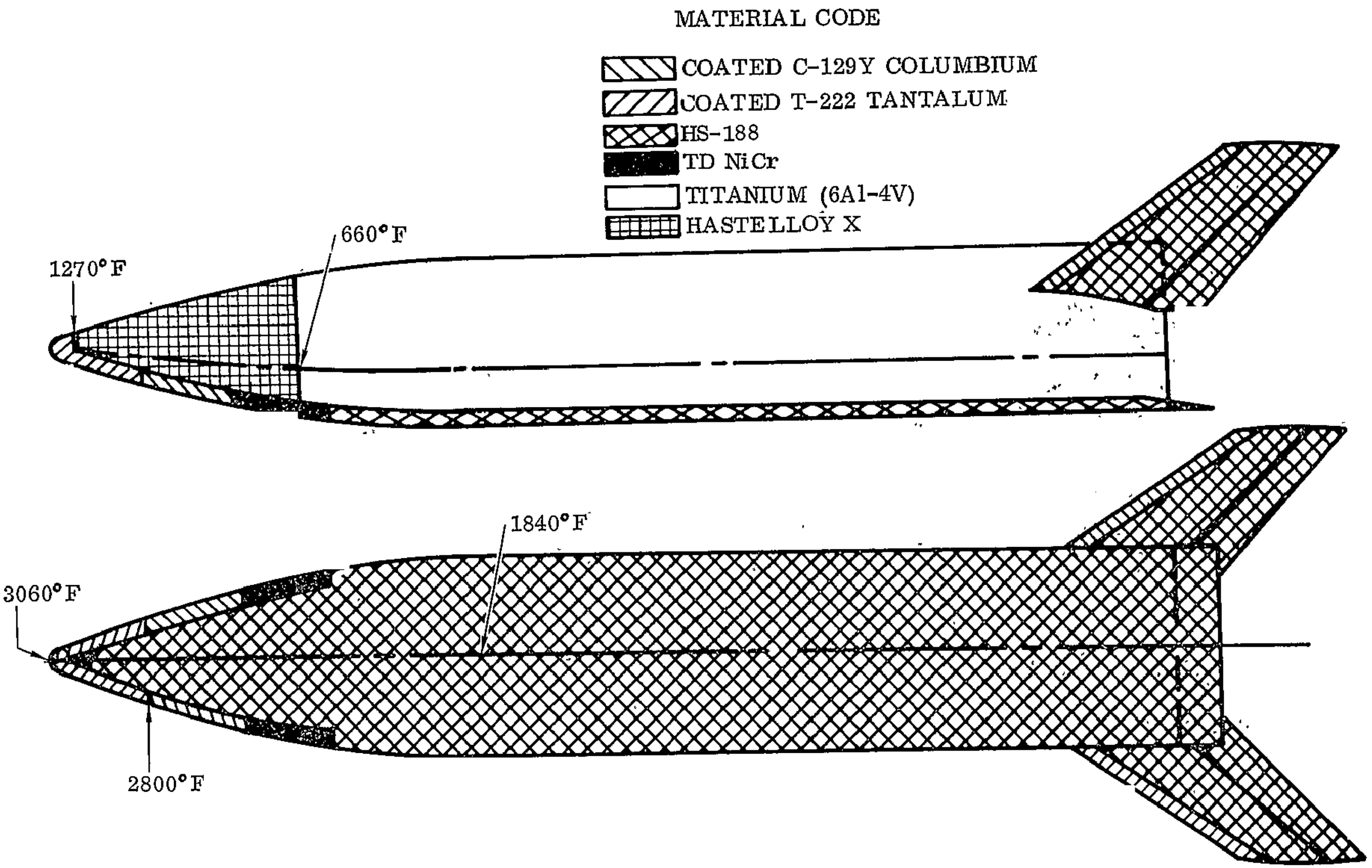

MATERIAL CODE
▧ COATED C-129Y COLUMBIUM
▨ COATED T-222 TANTALUM
▩ HS-188
■ TD NiCr
☐ TITANIUM (6Al-4V)
▦ HASTELLOY X
1270°F
660°F
3060°F
1840°F
2800°F

Figure 8-31. Orbiter Heat Shield Materials and Design Temperatures for FR-3 and FR-4

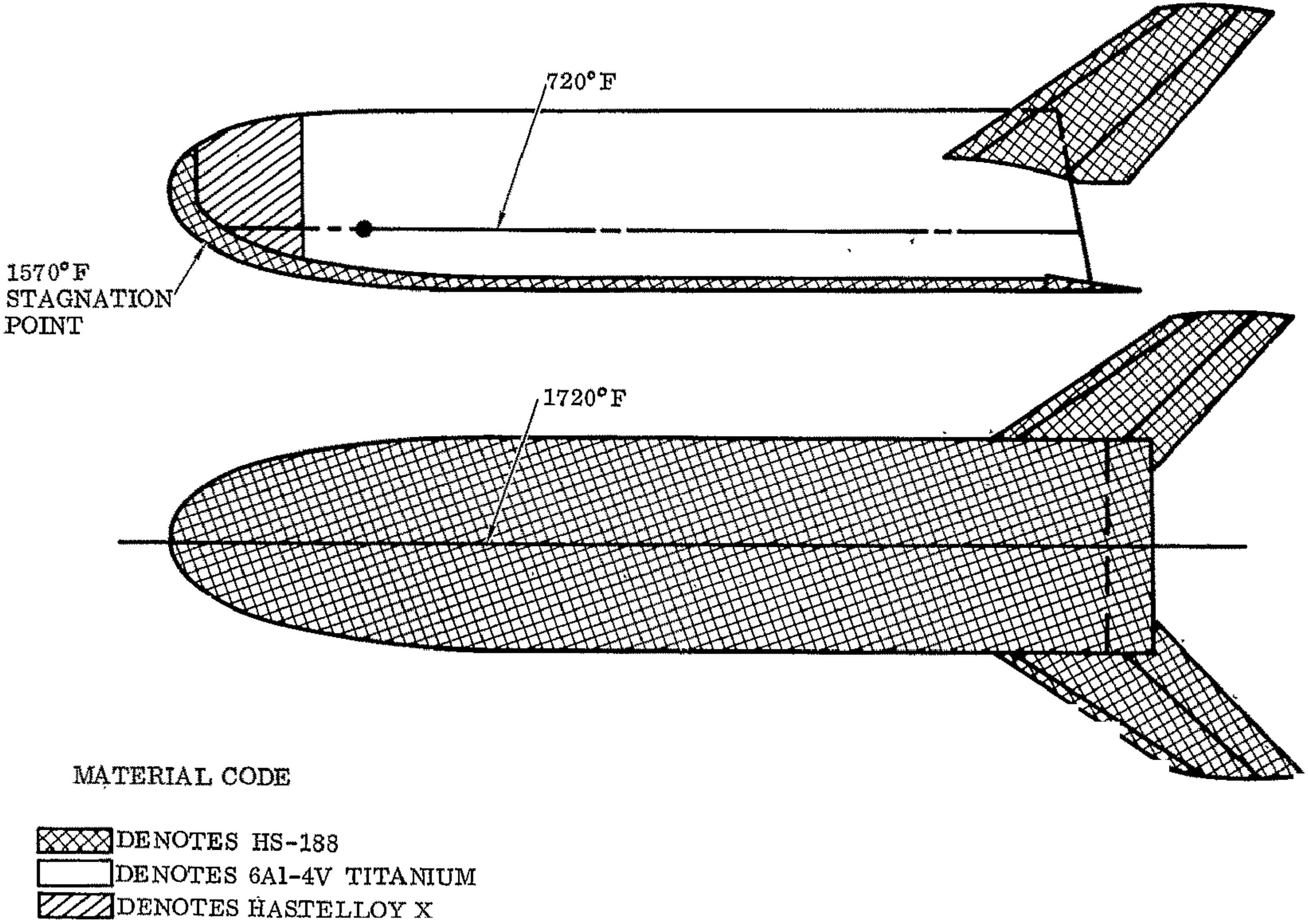
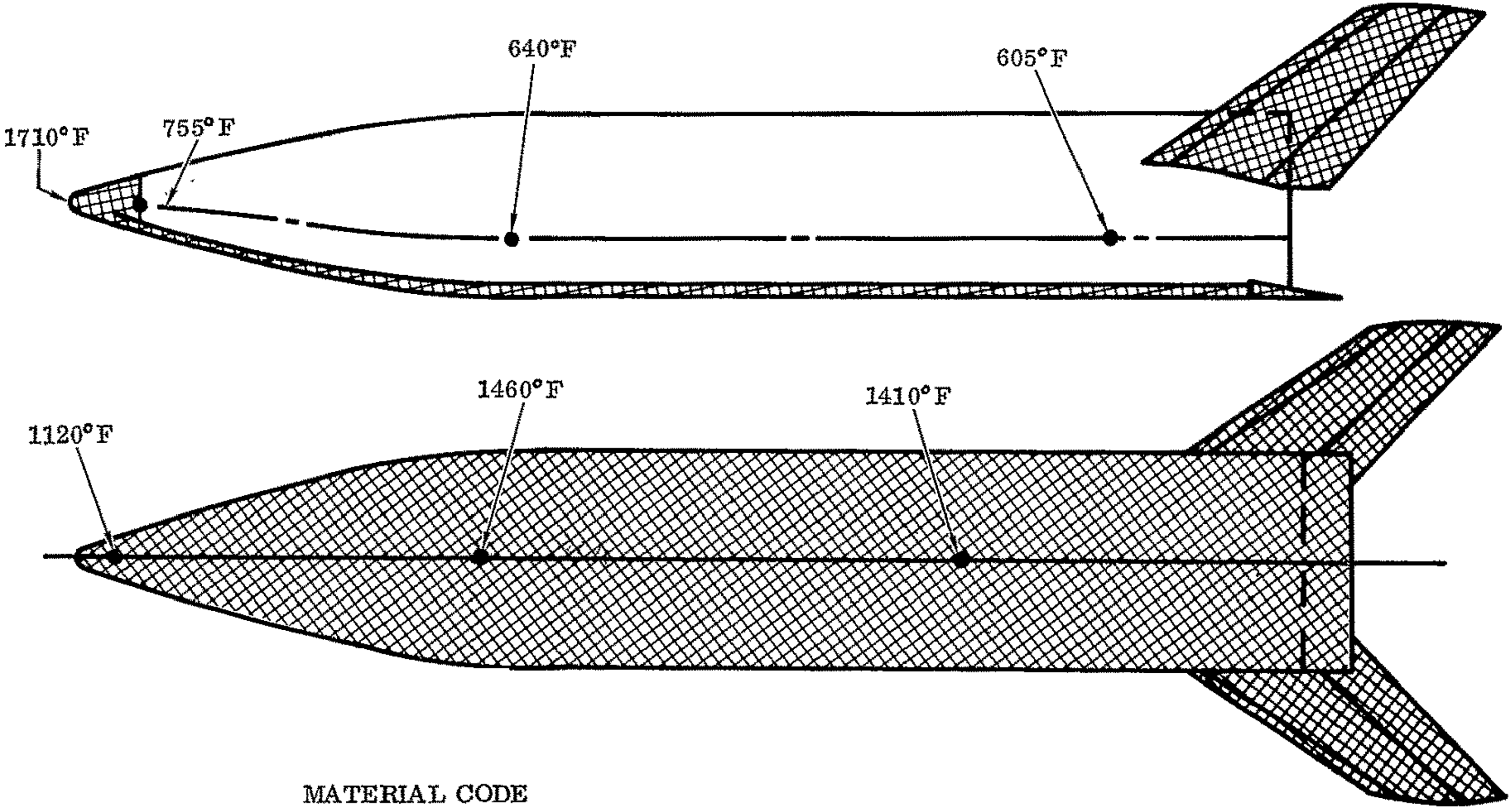

720° F
1570° F STAGNATION POINT
1720° F
MATERIAL CODE
▩ DENOTES HS-188
□ DENOTES 6A1-4V TITANIUM
▨ DENOTES HASTELLOY X

Figure 8-32. Booster Heat Shield Design Temperatures and Materials for FR-3


640°F
605°F
1710°F
755°F
1460°F
1410°F
1120°F

MATERIAL CODE

▨ DENOTES HASTELLOY X

☐ DENOTES 6Al-4V TITANIUM

Figure 8-33. Booster Heat Shield Design Temperatures and Materials for FR-4

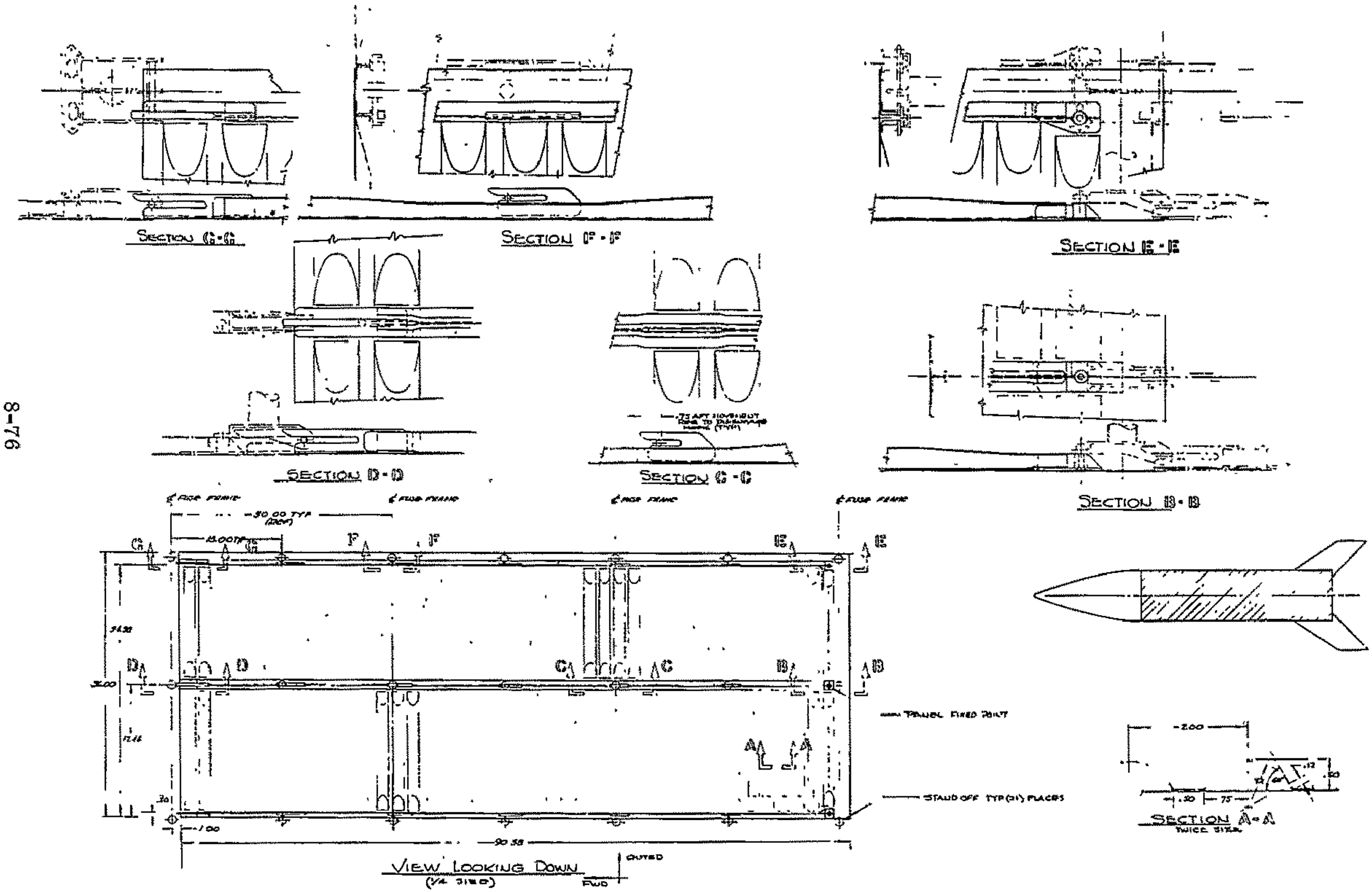

SECTION G-G
SECTION F-F
SECTION E-E
SECTION D-D
SECTION C-C
SECTION B-B
.75 AFT HOW BOLT
℄ FUSE FRAME
℄ FUSE FRAME
℄ FUSE FRAME
℄ FUSE FRAME
30.00 TYP
15.00 TYP
G
G
F
F
E
E
D
D
C
C
B
B
A
A
34.38
3.00
11.46
.30
1.00
90.38
PANEL FIXED POINT
STAND OFF TYP (21) PLACES
VIEW LOOKING DOWN
(1/4 SIZE)
OUTBD
FWD
2.00
.12
.50
.20
.75
SECTION A-A
TWICE SIZE

Figure 8-34. Typical HS-188 Heat Shield Panel

removing three screws, sliding the panel aft to disengage the hooks (permitted by the space provided for fore and aft thermal expansion), tilting up one free edge, then sliding the panel sideways to free. One long edge of each panel laps under an adjacent panel while the other long edge laps over its adjacent panel. Sheet metal extensions from two panel edges provide a sealing lip with the adjacent panels. Leakage area on the downstream edge to control the Δp is provided by chem-milling shallow grooves in the adjacent panel. These shallow grooves provide a leakage area in addition to that created by the laps of the sealing lips where four panels intersect. These grooves provide their additional leakage area only while the panels are hot.

The Hastelloy X heat shields are similar in design to the HS-188 heat shields except that the panels are supported on a grid approximately 18 × 30 inches instead of 18 × 15 inches. The increased grid spacing is permitted because the allowable strengths of Hastelloy X at its maximum use temperature are much higher than the strengths of HS-188 at its maximum use temperature. Hastelloy X is less dense than HS-188.

Figure 8-35 shows a rectangular panel fabricated from TD NiCr. It is made by diffusion bonding ribs and stiffening beams to a sheet of TD NiCr; thicknesses are then controlled by chem-milling. The method is generally adapted to brazing also if a wider braze joint is provided. The small areas where TD NiCr is used require that the shape be modified from the rectangular shape of Figure 8-35. However, the support grid will be approximately 15 × 18 inches. The methods of attachment, removal, sealing, and restraint are similar to those for the HS-188 heat shields.

The refractory metal alloys, T-222 tantalum and C-129 columbium, are used only on the orbiter for the nose and leading edges. The basic requirement for using the refractory metals at high temperatures is to provide a reliable coating to resist oxidation. The problem is alleviated somewhat since the heating occurs in a low oxygen pressure environment. Convair tests reported in GDC-ERR-1272 and GDC-ERR-1345 and continuing tests conducted since those reports show that reliable coatings are available. Convair considers a conservative life for T-222 to be 10 cycles to 3100°F and for C-129Y to be 35 cycles to 2500°F before replacement. T-222 tantalum and C-129Y nose and leading edge heat shields are made by welding rib stiffeners to formed sheet. Such a configuration, subsequently chem-milled, will provide fillets and good access to each corner. Fillets and good access facilitate application, resistance to service cracking, inspection, and repair of the coating.

Figure 8-36 shows a typical construction for 5 by 16-foot flat-side panels of 6Al-4V titanium. The basic material is Stresskin sandwich, a welded and diffusion bonded honeycomb widely used on the SST with good demonstrated acoustic resistance. Panels are stiffened by welded-on beams and braced by struts from the basic structure. Nine attach points support each panel. There is access to these panels from inside the vehicle.

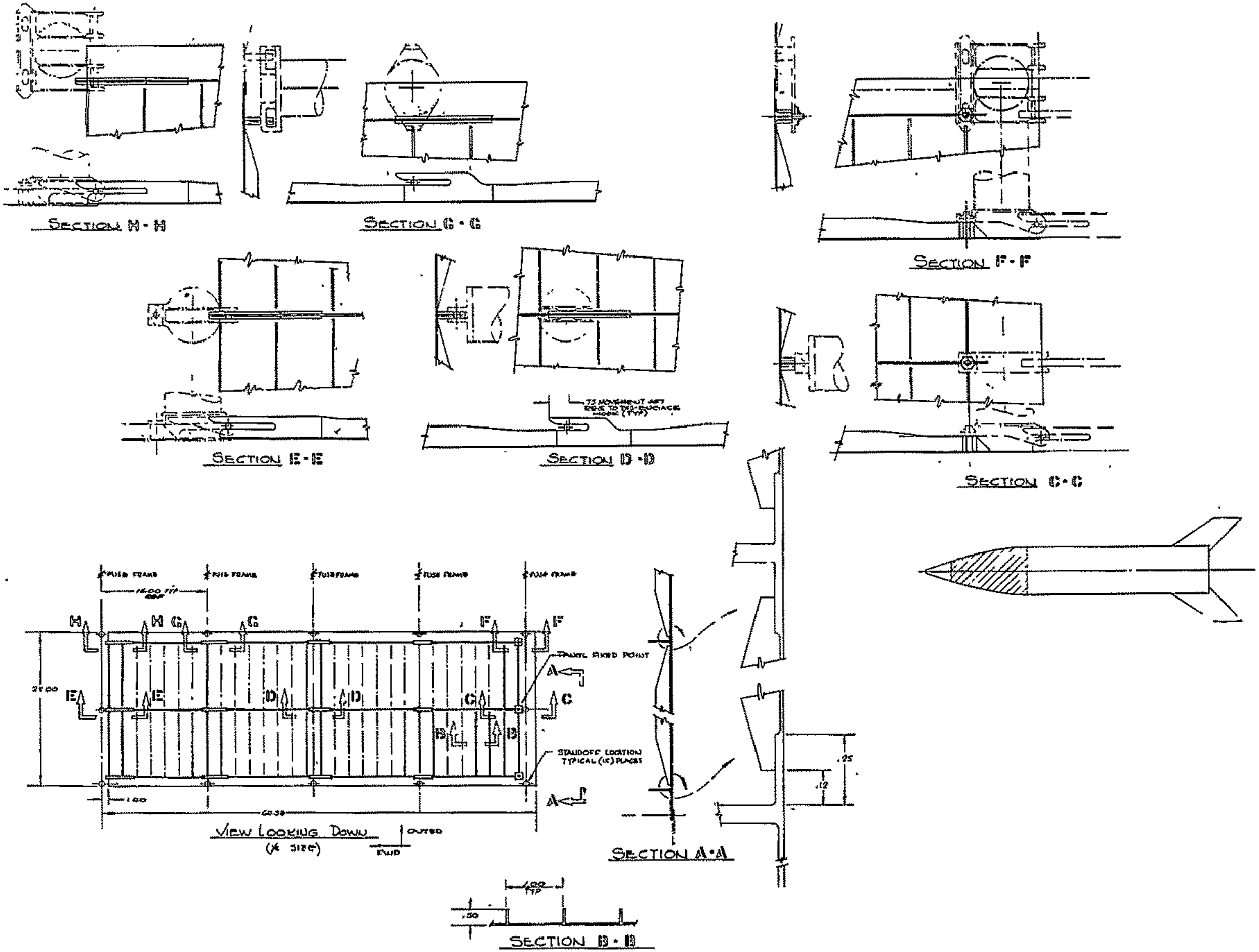

SECTION H-H
SECTION G-G
SECTION F-F
SECTION E-E
SECTION D-D
.75 MOVEMENT AFT REQ TO DIS-ENGAGE HOOK (TYP)
SECTION C-C
℄ FUSE FRAME
16.00 TYP REF
H
G
F
E
D
C
B
A
21.00
PANEL FIXED POINT
STANDOFF LOCATION TYPICAL (12) PLACES
1.00
60.38
VIEW LOOKING DOWN
(¼ SIZE)
OUTBD
FWD
SECTION A-A
.25
.12
4.00 TYP
.50
SECTION B-B

Figure 8-35. Typical TD NiCr Heat-Shield Panel

The tanks are joined to adjacent structural sections of the vehicle by means of a bolted butt joint at the point of tangency of the tank dome and skin. Their design incorporates fittings, doublers, and other reinforcements to introduce and redistribute concentrated loads such as turbofan engine thrust and vehicle interconnect and staging forces.

The intertank adapter structure is located between the LO_2 and LH_2 tanks in the constant section of the vehicle. Its structural skins are stiffened by longitudinal stringers and fuselage rings and provides a similar bolted tank attachment.

The aft end of the LH_2 tank is joined to the boost engine thrust skirt which supports a matrix of deep-section beams. These intersecting beams support 15 gimbal pads and transfer the main engine thrust loads into the skirt skin. The thrust skirt also incorporates a center tie box for the stabilizers. Beneath the thrust skirt and in line with the flat bottom of the vehicle, a multicell beam structure extends from side to side. Two holddown fittings, one on either side, are attached to this beam. A third holddown fitting is mounted to the upper end of the vertical main engine thrust beam.

The vehicle employs a conventional deployable-wing design with "reverse" type flaps at the trailing edge. The wing box has two spars, ribs, and skins with integral stringers. The wing pivot fitting is a fusion-welded assembly of titanium alloy. Leading edge and fixed trailing edge assemblies are rib-stiffened contoured skins and honeycomb sandwich panels, respectively. The panels are supported on beams cantilevered from the rear spar.

Stabilizer construction of the booster differs from that of the orbiter because of a different thermal environment. The stabilizer of the booster is a hot structure throughout, made entirely from 718 nickel-base alloy and incorporating a two-spar box beam with rib and stringer stiffened skins. Leading edge and movable trailing edge surfaces are rib-stiffened contoured skin assemblies.

The TPS that covers the exterior of the entire vehicle, with the exceptions mentioned earlier, is mounted to its support structure and does not contribute to the ability of the basic vehicle structure to resist external and internal forces. The beams, membranes, and braces of the support structure are attached to the vehicle structure and transfer aerodynamic loads only.

An alternate structural arrangement for the FR-3 booster is shown in Figure 8-40. This arrangement presents a "hot" structure approach where the heat shield has been stiffened and supported with frames to carry the primary flight loads. The propellant tanks are installed within the airframe so they are isolated from thermal loads and deflections.

8.6 COMPOSITE MATERIALS APPLICATION

Research into structural materials for aerospace use has recently concentrated on filamentary composites. Of the large number of possible matrix-filament combinations, the most advanced currently are those using boron filament with either epoxy resin or diffusion-bonded aluminum matrix materials.

Aluminum-boron (Al-B) composite material exhibits good creep and fatigue strength up to 800°F, making it a candidate for launch and reentry vehicles where titanium is now used. Al-B is 3.5 times as stiff as aluminum and twice as strong, offering significant weight savings for lower temperature aluminum airframes.

During the past three years, Convair has evaluated 26 different metal matrix composite materials including boron, silicon carbide, graphite, metallic wires, whiskers, and coated filaments in combination with aluminum, titanium, and nickel matrices.

The properties of most interest to the designer in the application of composite materials to aerospace structural hardware include availability, cost, tensile (compression), fatigue properties, and fabricability. A summary of these properties as well as information on nondestructive testing, specifications, notched toughness, and crack propagation properties; on creep, elevated, and low temperature mechanical properties; and on physical properties, corrosion, and compatibility may be found in References 8-4 through 8-11.

A program has also been performed to develop and evaluate acceptable means for fabricating structural hardware from Al-B composite material. Techniques have been developed for successfully roll forming and bending Al-B composite sheet material in both unidirectional and cross-ply material. At elevated temperatures, sheet material has been successfully bent to a 4.5-foot bend radius. Stringers and longerons are commonly used structural members which are fabricated by roll forming.

The joining of Al-B to itself and to other structural materials has been most successful. One of the most promising methods is resistance welding (spot and roll seam), which results in very high joint strengths. Other joining methods include mechanical joints, brazing, and adhesive bonding. Such techniques are normally used in joining Al-B caps to a titanium spar structure or joining Al-B stringers or stiffeners to a titanium web.

The Al-B composite is a highly anisotropic material with very high strength and stiffness along the filament direction and significant ductility across the filament direction. To utilize the material fully, the filaments in the composite can be oriented in the direction of load to match the material properties.

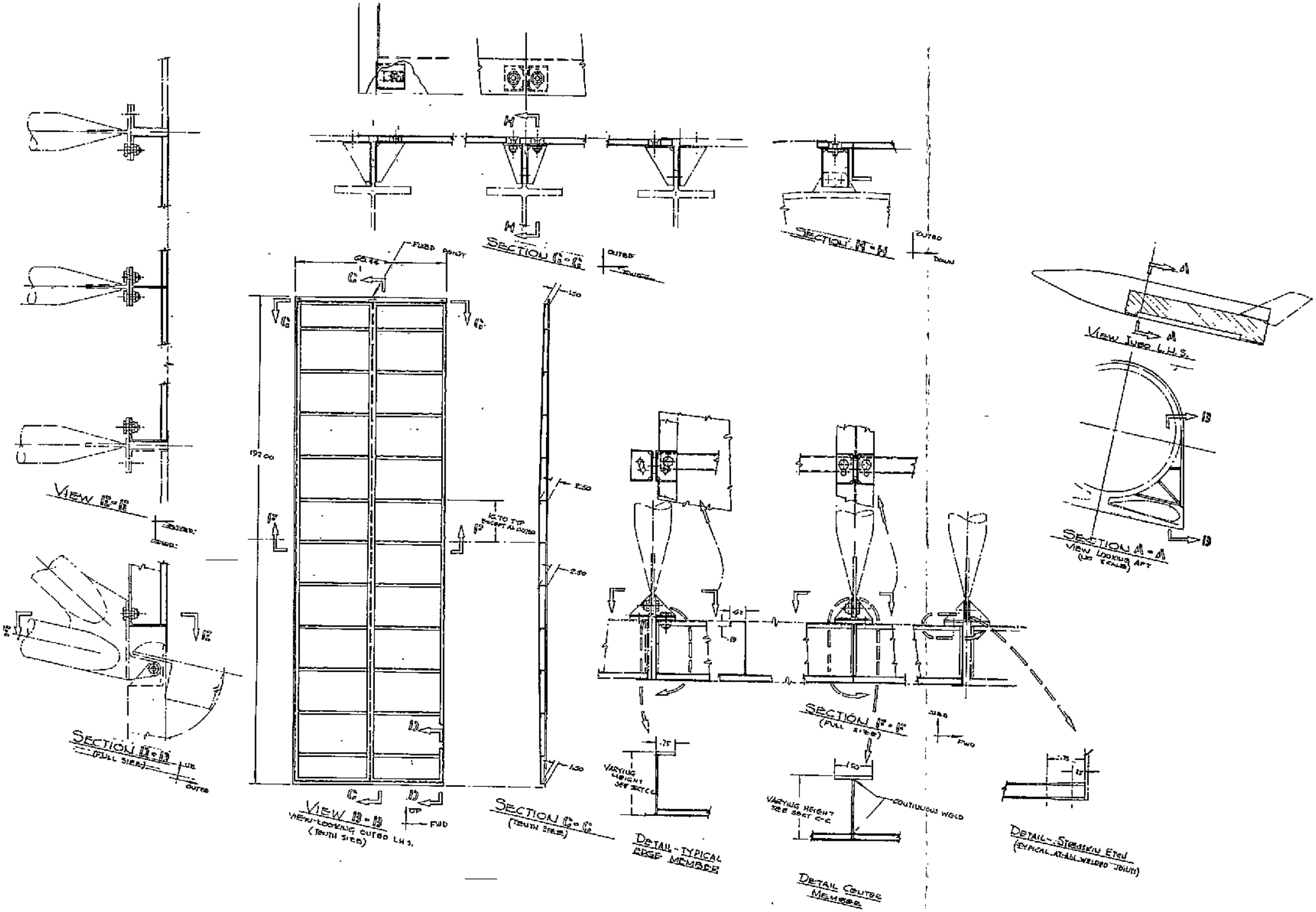

View E-E
Section D-D
(Full Size)
View B-B
View Looking Outbd LHS
(Truth Size)
192.00
Section C-C
(Truth Size)
Section G-G
Section H-H
Detail - Typical Edge Member
Varying Height See Sect C-C
Section F-F
(Full Size)
Detail Center Member
Continuous Weld
Detail - Stressskin Edge
(Typical at all welded joints)
View Inbd L.H.S.
Section A-A
View Looking Aft
(No Scale)
FWD
Outbd

Figure 8-36. Titanium Heat Shield (Flat Panel)

Contoured 6Al-4V Ti panels for the top of the vehicle are shown in Figure 8-37. Attachments are made directly to fittings spaced at about 20 inches circumferentially on the basic structural frames, which have a 30-inch fore and aft spacing. The attachment, removal, and relative motions are similar to the approach used for the HS-188 panels.

A typical heat shield support post is shown in Figure 8-38. It is basically a thin-walled cylinder with a fitting at the outer end (to which the heat shields are attached) and a provision at the base for stable attachment to the TPS support structure. Even when hot, the outer end of the cylinder must be able to carry the 0.1 psi heat shield load. The cold end must carry the moment developed by a 40 g vibratory load from the heat shield mass. Within these load-carrying requirements, the cylinder wall is made as thin as feasible to minimize heat conduction along the length of the post. The inside of the cylinder is packed with fibrous insulation to block heat transfer in that area. If necessary, the heat transfer through the post can be reduced by using a thinner wall, higher strength material such as Rene' 41 for as much of the cylinder length as temperature limits permit. For very thin walls, such as 0.004 to 0.006 in., the cylinder would have to be stabilized against buckling by such methods as corrugating the wall and/or adding circumferential stiffening rings. As shown by Figure 8-38, the material for the outer end fitting of the post is usually the same material as the heat shield that it supports. This outer end fitting reaches as far down the post as necessary to reach the allowable temperature of the next material.

To provide longer life with equal or greater reliability, certain backup materials may be considered for heat shields instead of coated tantalum, coated columbium, and possibly TD Ni Cr. The principal backup material candidates are coated carbon-carbon composite and zirconium diboride. The total area of heat shields using the coated refractories is less than 100 ft^2 on the orbiters and zero on the boosters. Therefore, the weight penalty would be small even if unit weights were higher. These possibilities are discussed further in Volume X, Section 4.

8.5 FR-3 BOOSTER THERMOSTRUCTURAL CONCEPT

The FR-3 booster employs two propellant tanks of 33 feet in diameter which are of similar construction as the Saturn S-IC stage. The forward LO_2 tank is 59.8 feet in length and separated from the aft LH_2 tank of 117.3 feet by a conventional semimonocoque structure. Separate tanks rather than a single divided tank were used in this configuration because the weight difference (2.6% of GLOW) was a reasonable tradeoff for the advantages of increased operational reliability and reduced development uncertainties inherent in the separate tank design.

The major structural assemblies shown in Figure 8-39 are:

a. Forward fuselage, including the crew compartment, turbofan engine bays, and nose landing gear.

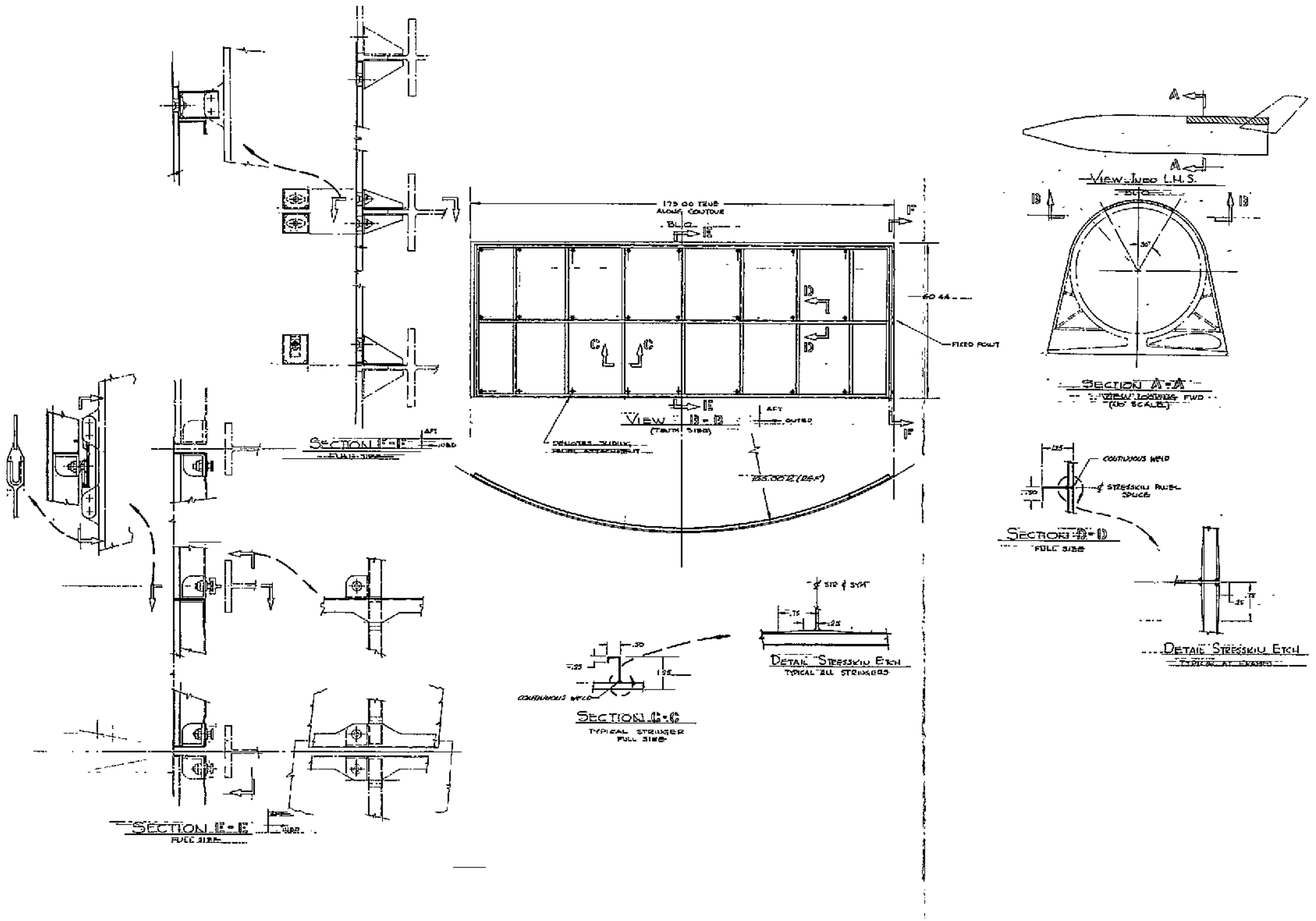

A
A
VIEW INBD L.H.S.
B
B
30°
SECTION A-A
VIEW LOOKING FWD
(NO SCALE)
175.00 TRUE ALONG CONTOUR
BLQ.
E
E
F
F
D
D
C
C
60.44
FIXED POINT
VIEW B-B
(TRUTH SIZE)
AFT
OUTBD
DENOTES SLIDING PANEL ATTACHMENT
135.062 (REF)
SECTION F-F
FULL SIZE
AFT
INBD
SECTION E-E
FULL SIZE
.125
CONTINUOUS WELD
℄ STRESSKIN PANEL SPLICE
.190
SECTION D-D
FULL SIZE
.25
.75
DETAIL STRESSKIN ETCH
TYPICAL AT FRAMES
℄ STR ℄ SYM
.75
.25
DETAIL STRESSKIN ETCH
TYPICAL ALL STRINGERS
.50
.25
1.25
CONTINUOUS WELD
SECTION C-C
TYPICAL STRINGER
FULL SIZE

Figure 8-37. Titanium Heat Shield (Contoured Panel)

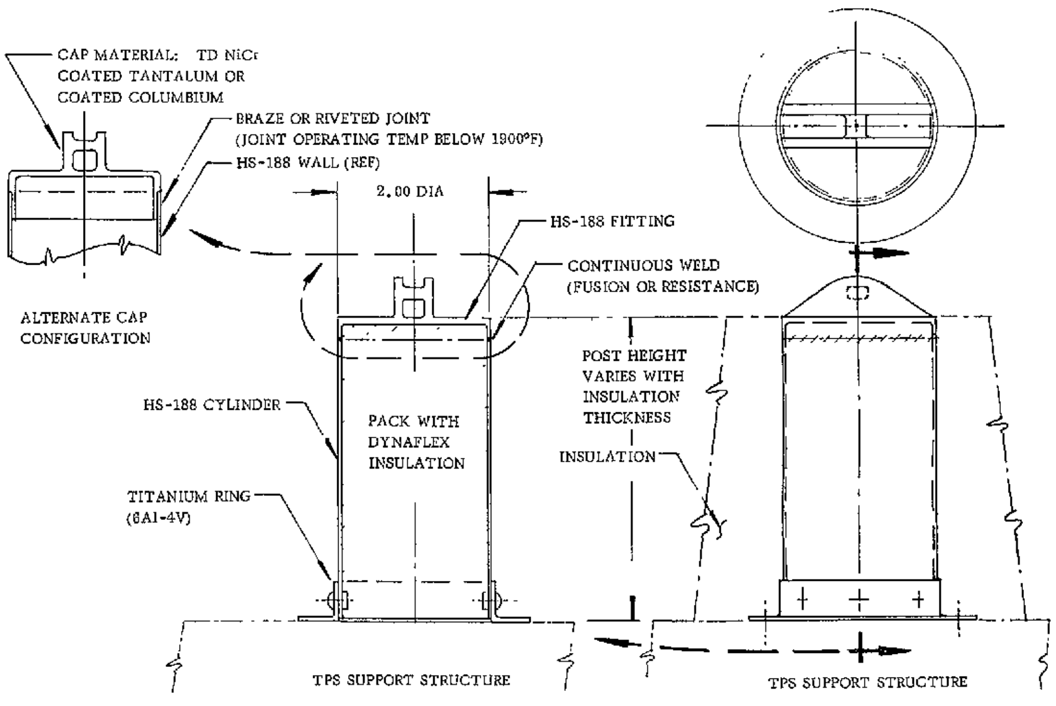

CAP MATERIAL: TD NiCr
COATED TANTALUM OR
COATED COLUMBIUM
BRAZE OR RIVETED JOINT
(JOINT OPERATING TEMP BELOW 1900°F)
HS-188 WALL (REF)
ALTERNATE CAP
CONFIGURATION
2.00 DIA
HS-188 FITTING
CONTINUOUS WELD
(FUSION OR RESISTANCE)
POST HEIGHT
VARIES WITH
INSULATION
THICKNESS
INSULATION
HS-188 CYLINDER
PACK WITH
DYNAFLEX
INSULATION
TITANIUM RING
(6Al-4V)
TPS SUPPORT STRUCTURE
TPS SUPPORT STRUCTURE

Figure 8-38. Typical Heat Shield Support Post

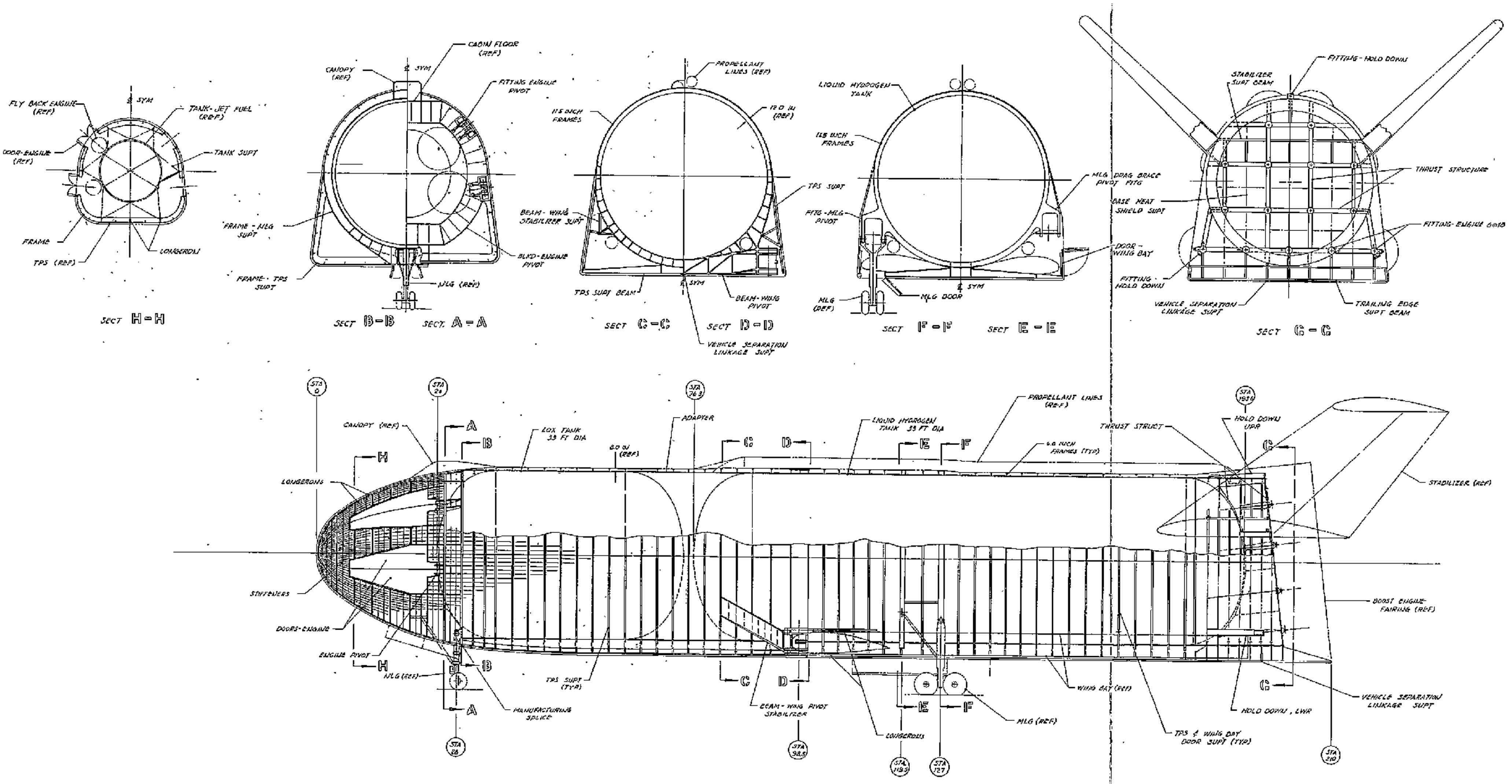

FLY BACK ENGINE (REF)
℄ SYM
TANK-JET FUEL (REF)
DOOR-ENGINE (REF)
TANK SUPT
FRAME
TPS (REF)
LONGERON
SECT H-H
CANOPY (REF)
℄ SYM
CABIN FLOOR (REF)
FITTING ENGINE PIVOT
FRAME - MLG SUPT
FRAME - TPS SUPT
BLKD-ENGINE PIVOT
MLG (REF)
SECT B-B
SECT A-A
PROPELLANT LINES (REF)
115 INCH FRAMES
12.0 IN (REF)
BEAM-WING STABILIZER SUPT
TPS SUPT
TPS SUPT BEAM
℄ SYM
BEAM-WING PIVOT
VEHICLE SEPARATION LINKAGE SUPT
SECT C-C
SECT D-D
LIQUID HYDROGEN TANK
115 INCH FRAMES
FITG-MLG PIVOT
MLG (REF)
MLG DOOR
℄ SYM
MLG DRAG BRACE PIVOT FITG
BASE HEAT SHIELD SUPT
DOOR-WING BAY
SECT F-F
SECT E-E
FITTING-HOLD DOWN
STABILIZER SUPT BEAM
THRUST STRUCTURE
FITTING-ENGINE GMB
FITTING-HOLD DOWN
VEHICLE SEPARATION LINKAGE SUPT
TRAILING EDGE SUPT BEAM
SECT G-G
STA 0
STA 24
STA 76.3
STA 193.5
CANOPY (REF)
A
B
H
LOX TANK 33 FT DIA
6.0 IN (REF)
ADAPTER
C
D
E
F
LIQUID HYDROGEN TANK 33 FT DIA
PROPELLANT LINES (REF)
6.0 INCH FRAMES (TYP)
THRUST STRUCT
HOLD DOWN UPR
G
STABILIZER (REF)
LONGERONS
STIFFENERS
DOORS-ENGINE
ENGINE PIVOT
MLG (REF)
MANUFACTURING SPLICE
TPS SUPT (TYP)
BEAM-WING PIVOT STABILIZER
LONGERONS
WING BAY (REF)
MLG (REF)
TPS & WING BAY DOOR SUPT (TYP)
HOLD DOWN, LWR
BOOST ENGINE FAIRING (REF)
VEHICLE SEPARATION LINKAGE SUPT
STA 26
STA 98.5
STA 119.5
STA 127
STA 210

Figure 8-39. Structural Arrangement, FR-3 Booster

b. LO_2 tank.

c. Intertank adapter section.

d. LH_2 tank, including wing pivot bulkhead, and landing gear bulkheads.

e. Thrust structure, including vehicle holddown supports, base heat shield support, stabilizer carry-through structure and vehicle separation supports.

f. Stabilizers and wings.

g. TPS and support structure.

Most structures, including the LO_2 and LH_2 tanks, have been designed for a thermal environment of 200° F or less. This temperature is maintained at the structural envelope during reentry by the TPS. The exceptions are the aerodynamic stabilizer and wing. During reentry, the deployable wing is stowed in its compartment and is not subjected to elevated temperatures. The stabilizers are designed from titanium alloy that can withstand elevated temperatures and maintain their structural integrity at reduced but acceptable levels.

The fuselage section forward of the LO_2 tank is a semimonocoque shell that contains the crew compartment, equipment bay, turbofan engine compartments, and nose landing gear. Whereas the crew compartment is designed to be pressurized, the remainder of the forward fuselage is vented to ambient conditions. A major bulkhead at Station 24 supports the turbofan engine pivots; one at Station 29 supports the nose landing gear. The design of this fuselage section adheres to the classical methods of shell stiffening through the use of frames, bulkheads, and stiffeners. Longerons have been arranged to carry and redistribute concentrated loads that occur in the vicinity of the cockpit canopy, the turbofan engine doors, and nose landing gear.

The LO_2 and LH_2 tanks are both designed to form an integral part of the load-carrying vehicle structure. They are fusion-welded assemblies of wide circular rings that form the tank skin and frames. At each end, ellipsoidal domes form the closure bulkheads. Major external frames in the LH_2 tank area are the wing pivot frame and the main landing gear frame. Through these areas the body was deepened to provide additional frame depth for increased load restraint.

Major wing support frames are located at Stations 98.5 and 82.5 and with interconnecting stabilizer beams provide shear and moment restraint for the wings through the wing pivot fittings.

Main landing gear support frames are located at Stations 120 and 127.5, which restrain the main landing loads and drag loads respectively.

Table 8-9. Candidate Composite Materials Application, Contd

Area	Station	Maximum Temperature	Structural Member	Parent Material	Aluminum-Boron Composite Usage
Wing-Folding	105.25	200°F	Wing Pivot Fitting & Support Bulkhead	Ti 8-1-1	Beam Caps & Stiffeners
			Wing Box	Ti 8-1-1	Beam Caps & Stringers
			Leading & Trailing Edge	Al Aly	----
LH_2 Tank Area	142.5-176.2	0-200°F	Tank Shell & Stiffeners	2219 Al Aly	----
			Frames	2219 Al Aly	Frame Flanges
Thrust Structure	176.2-189.5	500°F	Thrust Skirt	Ti 8-1-1	Stringers & Frames
			Thrust Support Beams	Ti 8-1-1	Caps & Stiffeners
			Holddown Structure	Ti 8-1-1	----
Vehicle Separation Support	177.5, 179.5, 189.5	200°F	Stage-Interconnect Support	Ti 8-1-1	Frame Caps, Longerons
			Base Heat Shield Support	Ti 8-1-1	----
Stabilizer	179.5-189.5				
Booster		1200°F	Total Structure	718 No Insul.	----
Orbiter		200°F	Stabilizer Box	Ti 8-1-1	Beam Caps & Stringers
Trailing Edge	189.5-207	500°F	Upper Skin, Frames	Ti 8-1-1	Beam Caps & Formers

Typical components of the orbiter vehicle that are considered candidates for Al-B composite fabrication are shown in Table 8-10. Redesigned components included spar caps, beam and frame caps, and stringers and longerons that are used throughout the wing, fuselage, and fin areas. One of the most significant weight savings was found in the bulkhead application. As an example, a bulkhead using unidirectional Al-B caps over a forged titanium carrier provides a 43-percent weight saving. Similar results are obtained in the wing, again using unidirectional caps of Al-B with a forged titanium carrier. The combination of titanium with its high shear capacity for the webs and unidirectional Al-B with its high strength and stiffness for the caps results in 42 percent spar weight waving. Titanium was selected for the shear carrier because of its very high shear capacity and bearing strength (for local attachment loads) and the similarity of its coefficient of thermal expansion to that of Al-B. It also offers the advantage of good elevated-temperature strength.

8.7 REFERENCES

8-1 Evaluation of Superalloys for Hypersonic Vehicle Honeycomb Heat Shields, R. K. Malik and A. R. Stetson, Technical Report AFML-TR-68-292, October 1968.

8-2 Oxidation Resistant Materials for Transpiration Cooled Gas Turbine Blades, Part I, Sheet Specimen Screening Tests, F. W. Cole, et al, NASA CR-930, February 1968.

8-3 Radiative Thermal Protection Systems Development for Maneuverable Reentry Spacecraft, W. E. Black, Convair report GDC-ERR-1272, February 1969.

8-4 Evaluation of the Structural Behavior of Filament-Reinforced Metal Matrix Composites, W. H. Schaefer, et al, Quarterly Progress Reports 1-5 on Contract F3361567-C-1548, July 1967-July 1968.

8-5 Development of NDT Methods and Techniques for Evaluation of Composite Materials, R. T. Anderson and K. Perun, Convair report GDC-ERR-AN-1204, December 1967.

8-6 "Resistance Welding of Metal Matrix Composites", M.S. Hersh, presented at AWS National Fall Meeting, Cincinnati, Ohio, October 7-10, 1968; published by AWS.

8-7 Application of Composite Materials to Ramjet Inlet Structures, R. J. Taylor, et al, AFML-TR-68-85, Vol. 1, May 1968.

8-8 Research on Boron Filament/Metal Matrix Composite Materials, E. G. Wolff and R. J. Hill, AFML-TR-67-140, June 1967.

8-9 *Development and Evaluation of the Diffusion Bonding Process as a Method to Produce Fibrous Reinforced Metal Matrix Composite Materials,* A. Toy, D. G. Atteridge, and D. I. Sinizer, AFML-TR-66-350, November 1966.

8-10 *Application of Advanced Fibrous Reinforcement Composite Materials,* C. H. Conliffe and A. S. Kjelby, AFML-TR-68-86, May 1968.

8-11 *Research and Development of an Advanced Composites Technology Base and Component Fabrication Program For Gas Turbine Compressor and Fan Blades,* M. Herman, et al, 1st yearly report under Contract AF33(615)-5321, July 1967.

SECTION 9

LANDING GEAR

9.1 REQUIREMENTS

9.1.1 GENERAL

a. 10,000-foot heavy-load ZI class airfield.

b. Landing limit ground reaction factor of 2 for a sinking speed of 12 fps.

c. Steering angle of 45 degrees minimum each side of neutral.

d. Ambient temperature environment of -65°F to +200°F.

e. Shock strut life of 1000 landings. Wheel, brake, and tire life of 100 landings.

f. Follow military specifications for design of:

1. Shock absorbers: MIL-L-8552
2. Steering system: MIL-S-8812
3. Wheels and brakes: MIL-W-5013
4. Tires: MIL-T-5041
5. Brake system: MIL-B-8584
6. Brake antiskid system: MIL-B-8075
7. Loads: MIL-A-8860 series

9.1.2 BOOSTER

a. FR-3: Design landing weight of 517,286 pounds at a maximum landing velocity of 180 knots.

b. FR-4: Design landing weight of 324,789 pounds at a maximum landing velocity of 175 knots.

9.1.3 ORBITER

a. FR-3: Design landing weight of 286,655 pounds at a maximum landing velocity of 170 knots.

b. FR-4: Design landing weight of 322,475 pounds at a maximum landing velocity of 175 knots.

9.1.4 COMMON HARDWARE. Common hardware can be used on the FR-4 booster and orbiter because the landing weights and velocities and gear loads are comparable.

9.2 BASELINE CONFIGURATION

9.2.1 GENERAL DESCRIPTION. The landing gear system for the FR-3 orbiter and the FR-4 booster and orbiter consists of two four-wheel bogie type main gears and two dual-wheel conventional nose gears. The FR-3 booster system consists of two four-wheel bogie type main gears and one dual-wheel conventional nose gear.

Each main and nose landing gear consists of a conventional air-oil telescoping shock strut, a folding drag brace, tires, wheels, and antiskid brakes. The nose gears also incorporate a steering system.

The geometry of each gear provides for a drag brace locking mechanism that acts as both the uplock and downlock. Initial motion of the gear hydraulic retract actuator unlocks the locking mechanism.

9.2.2 SHOCK ABSORBERS. The landing gear shock absorbers are conventional air-oil type with a total stroke of 24 inches. The outer cylinder bore diameters are:

a. FR-3 booster MLG: 14 inches

b. FR-3 booster NLG: 6 inches

c. FR-3 orbiter MLG: 9 inches

d. FR-3 orbiter NLG: 6 inches

e. FR-4 booster and orbiter MLG: 10 inches

f. FR-4 booster and orbiter NLG: 6 inches

9.2.3 WHEELS AND TIRES. The wheels and tires were selected on the basis of:

a. Sizes used on present day aircraft.

b. Wheel loading.

c. Provisions for brake space.

d. Clearances with gear retracted.

e. Ground flotation capability for a heavy-load ZI class landing field.

9.2.3.1 FR-3 Booster. The MLG wheels and tires are size 56 × 16 inches with a twin center-to-center spacing of 36 inches and a tandem center to center spacing of 66 inches. The NLG wheels and tires are size 36 × 11 inches with a twin center-to-center spacing of 28 inches.

9.2.3.2 FR-3 Orbiter. The MLG and NLG wheels and tires are size 36 × 11 inches. The MLG twin center-to-center spacing is 30 inches and the tandem center-to-center spacing is 48 inches. The NLG twin center-to-center spacing is 28 inches.

9.2.3.3 FR-4 Booster and Orbiter. The MLG and NLG wheels and tires are size 44 × 13 inches. The MLG twin center-to-center spacing is 36 inches and the tandem center-to-center spacing is 60 inches. The NLG twin center-to-center spacing is 36 inches.

9.2.4 WHEEL BRAKES. The wheel brakes are conventional steel heat stack brakes installed on the MLG and NLG wheels. The wheel brake system is hydraulically actuated. Emergency operation is provided by a redundant backup system. These systems are completely independent and actuate separate sets of pistons in each brake. Metered pressure is available from either system.

The antiskid system is a fully modulated system with individual wheel control to develop maximum safe braking drag and is available when operating with either hydraulic system.

The wheels will accept brakes with the new lightweight beryllium and carbon heat stack materials when they are fully developed. The brake weight can be reduced approximately 50 percent by the use of these materials for the heat stack and titanium for the structural parts.

A 10,000-foot runway is not adequate for stopping with only the wheel brakes when landing over a 50-foot obstacle on a wet pavement ($\mu = 0.17$). A 50-foot-diameter drag chute is provided for deployment at touchdown for low friction coefficient pavement conditions.

9.2.5 NOSE WHEEL STEERING. A hydraulic actuated nose wheel steering unit is installed on each NLG. The steering units also provide shimmy damping.

9.2.5.1 FR-3 Booster. A powered steering angle of 80 degrees each of neutral with a free swivel angle of 20 degrees beyond the steering angle is provided on the FR-3 booster without manual disconnect.

9.2.5.2 FR-3 and FR-4 Orbiter and FR-4 Booster. A powered steering angle of 45 degrees each side of neutral is provided for each of the dual-wheel nose gears.

The steering units are operated by a common electrohydraulic servo unit to assure that the output to each steering unit is the same for any pilot steering input.

For ground handling, the torque links can be disconnected to provide 45 degrees of swivel beyond the powered steering angle.

9.2.6 TURNOVER ANGLE. The maximum turnover angle allowed by AFSCM 80-1 is 63 degrees. Figure 9-1 shows a 56-degree turnover angle for the FR-3 booster. Figure 9-2 compares the FR-4 booster and orbiter turnover angle for one nose gear at Station 38 and the two-nose-gear configuration considered in this report. Figure 9-3 compares the FR-3 orbiter turnover angle for one nose gear at the flyback engine mount bulkhead and the two-nose-gear configuration considered in this report.

9.2.7 LANDING GEAR ARRANGEMENT. The FR-3 booster landing gear arrangement is the conventional tricycle type. The FR-3 orbiter and the FR-4 booster and orbiter arrangements are similar to the B-52 bomber and are shown in Figures 9-4 and 9-5. The B-52 also has wing tip gears that normally contact the ground only under maximum gross weight ground maneuvering conditions. The B-52H maximum takeoff weight is 488,000 pounds. The design landing speed is 120 knots at a design landing weight of 270,000 pounds and a sink speed of 8 feet per second.

9.3 SYSTEM WEIGHT

The estimated system weights using present state-of-the-art materials and component designs are:

a. FR-3 booster: 25,000 pounds

b. FR-3 orbiter: 13,000 pounds

c. FR-4 booster: 16,000 pounds

d. FR-4 orbiter: 16,000 pounds

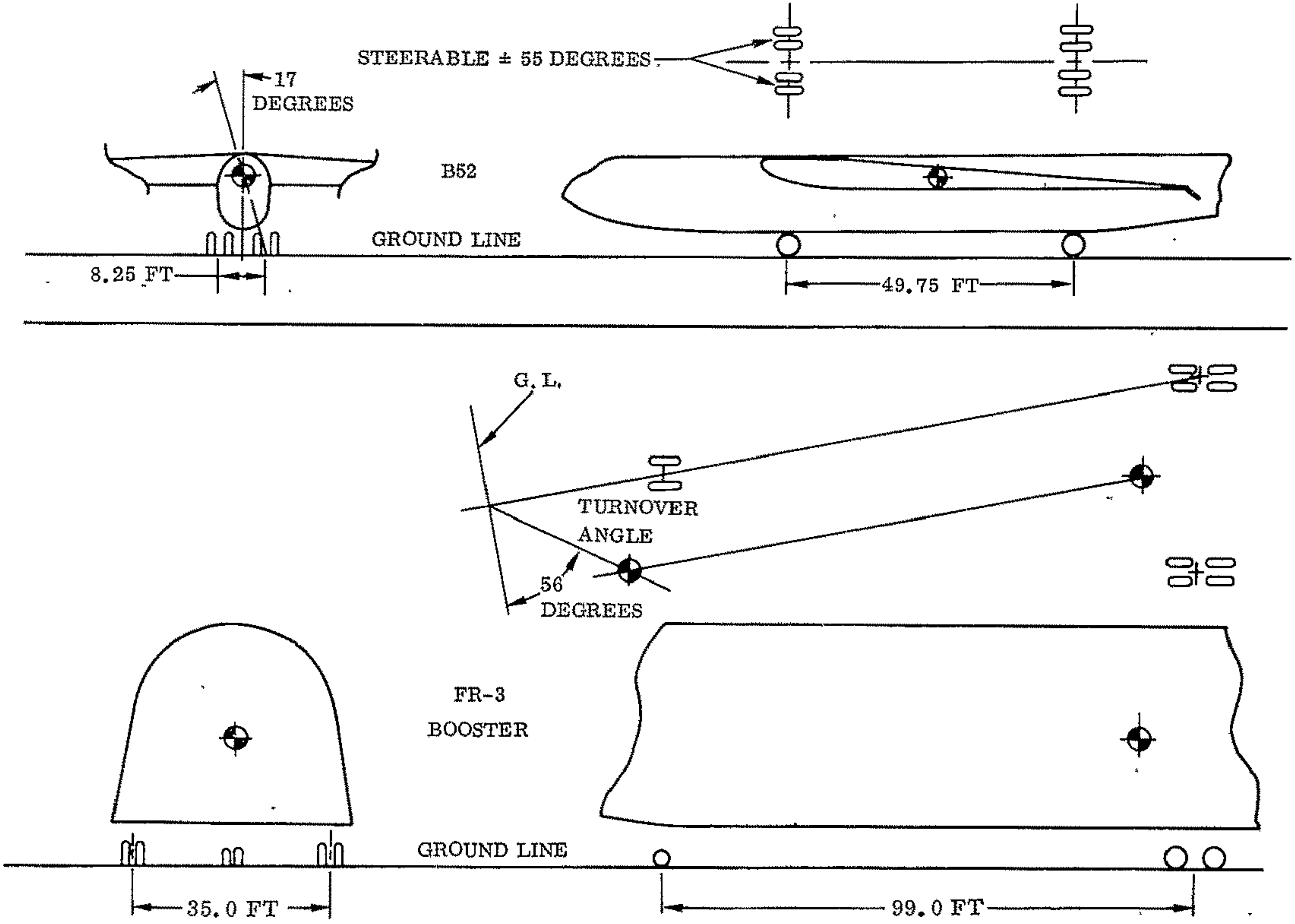

STEERABLE ± 55 DEGREES
17 DEGREES
B52
GROUND LINE
8.25 FT
49.75 FT
G. L.
TURNOVER ANGLE
56 DEGREES
FR-3 BOOSTER
GROUND LINE
35.0 FT
99.0 FT

Figure 9-1. 56-Degree Turnover Angle for FR-3 Booster

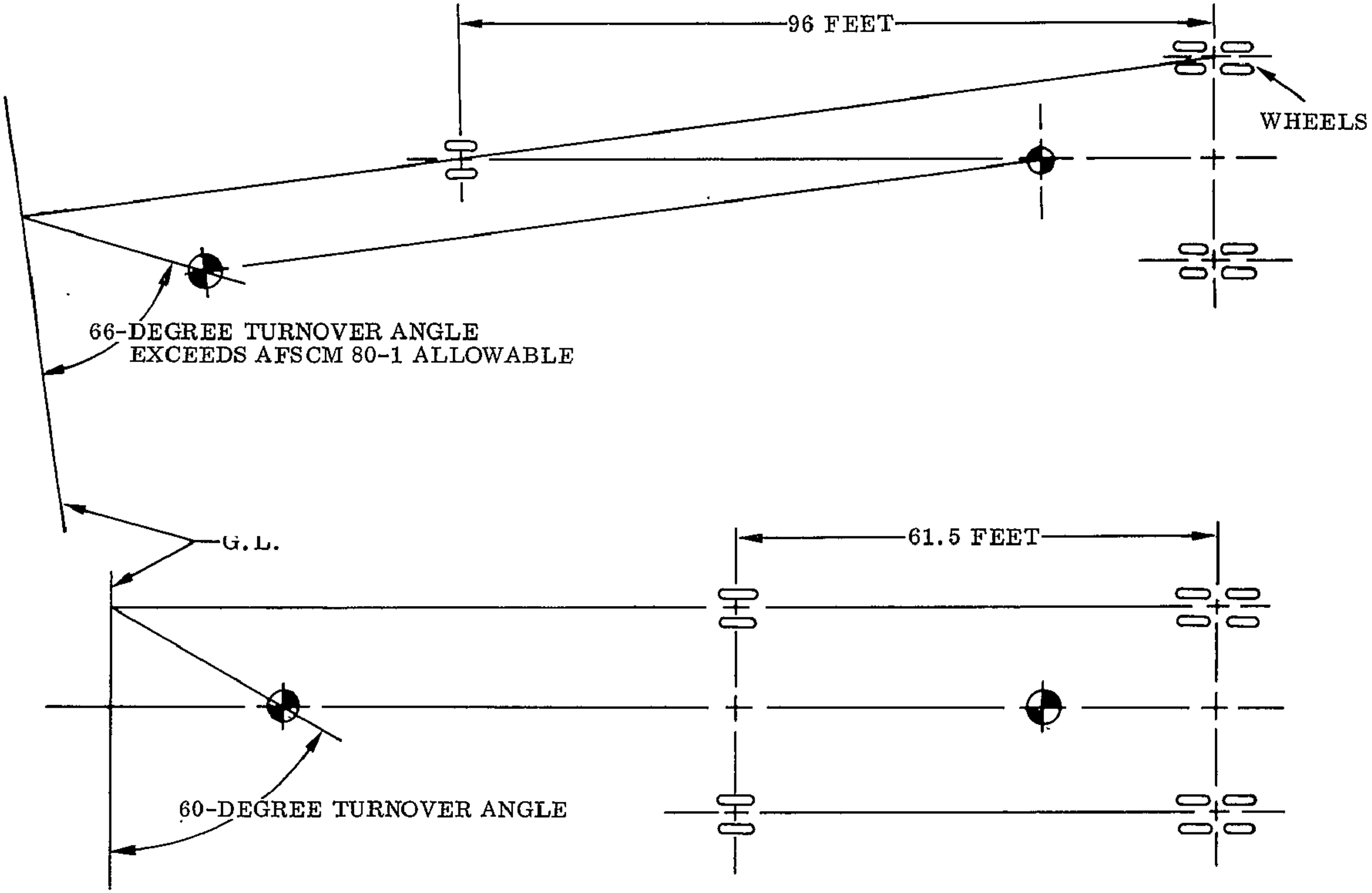

96 FEET
WHEELS
66-DEGREE TURNOVER ANGLE
EXCEEDS AFSCM 80-1 ALLOWABLE
G.L.
61.5 FEET
60-DEGREE TURNOVER ANGLE

Figure 9-2. FR-4 Booster and Orbiter Turnover Angle Comparison

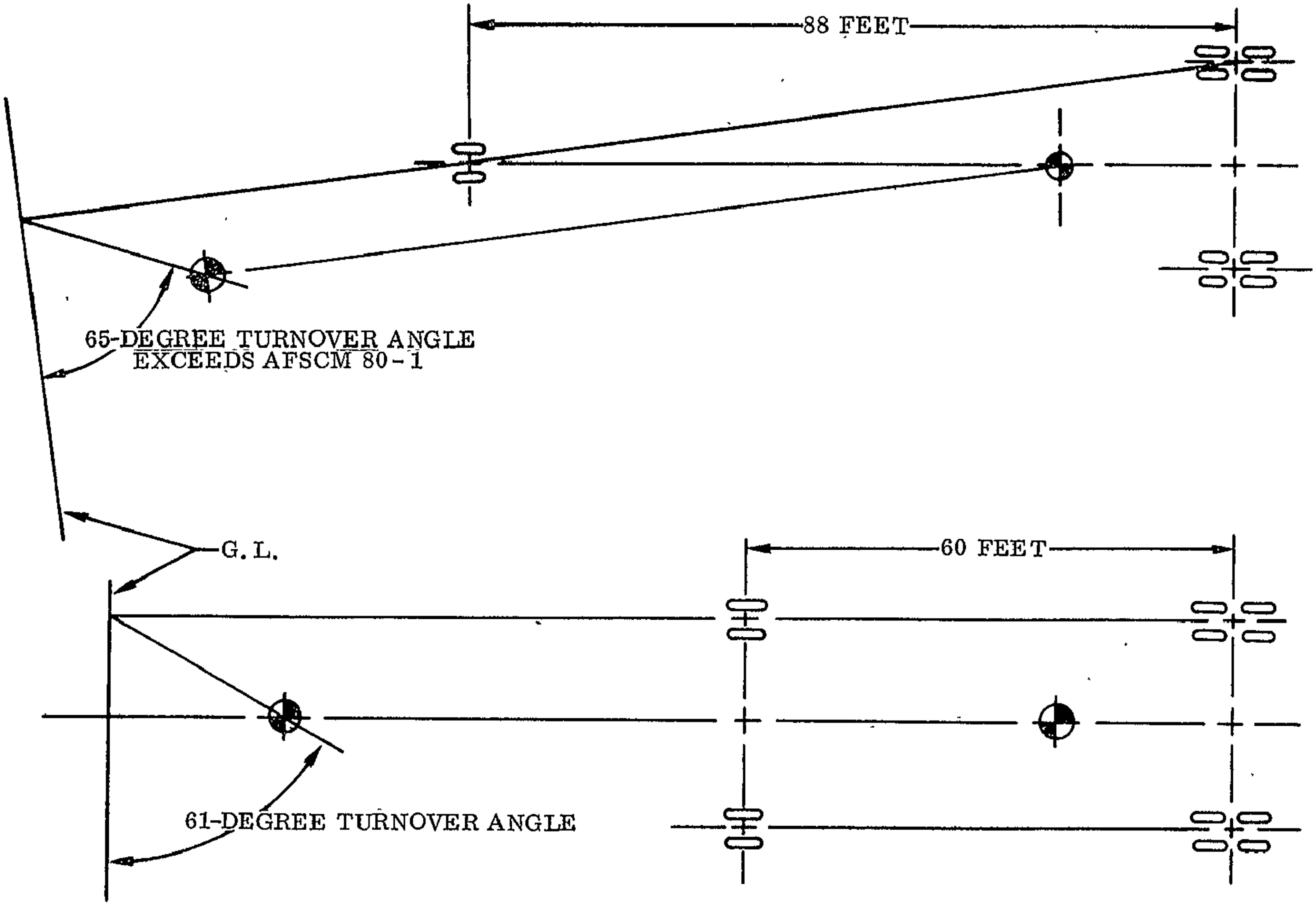

88 FEET
65-DEGREE TURNOVER ANGLE
EXCEEDS AFSCM 80-1
G. L.
60 FEET
61-DEGREE TURNOVER ANGLE

Figure 9-3. FR-3 Orbiter Turnover Angle Comparison

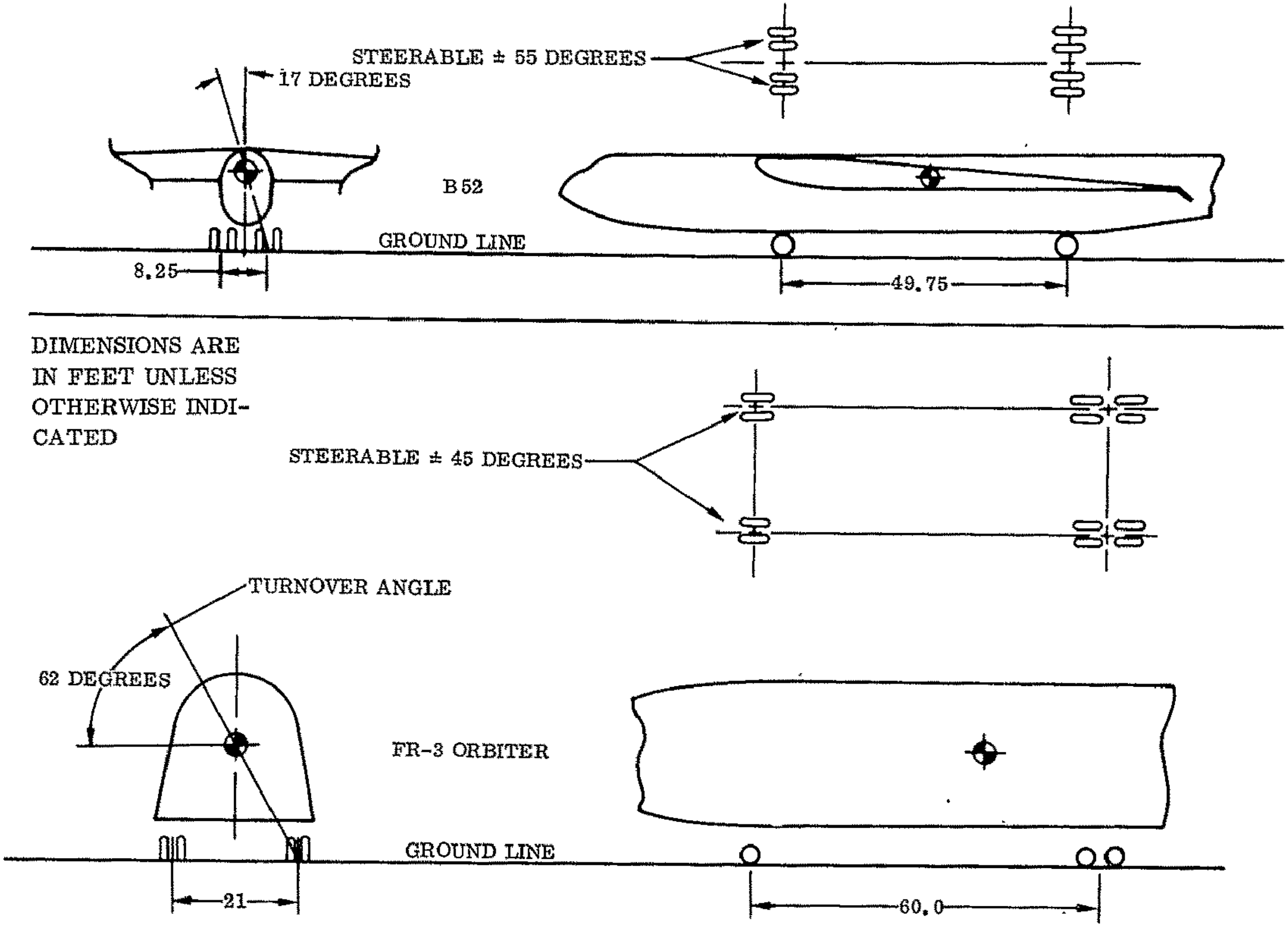

STEERABLE ± 55 DEGREES
17 DEGREES
B 52
GROUND LINE
8.25
49.75
DIMENSIONS ARE IN FEET UNLESS OTHERWISE INDICATED
STEERABLE ± 45 DEGREES
TURNOVER ANGLE
62 DEGREES
FR-3 ORBITER
GROUND LINE
21
60.0

Figure 9-4. Landing Gear Arrangement

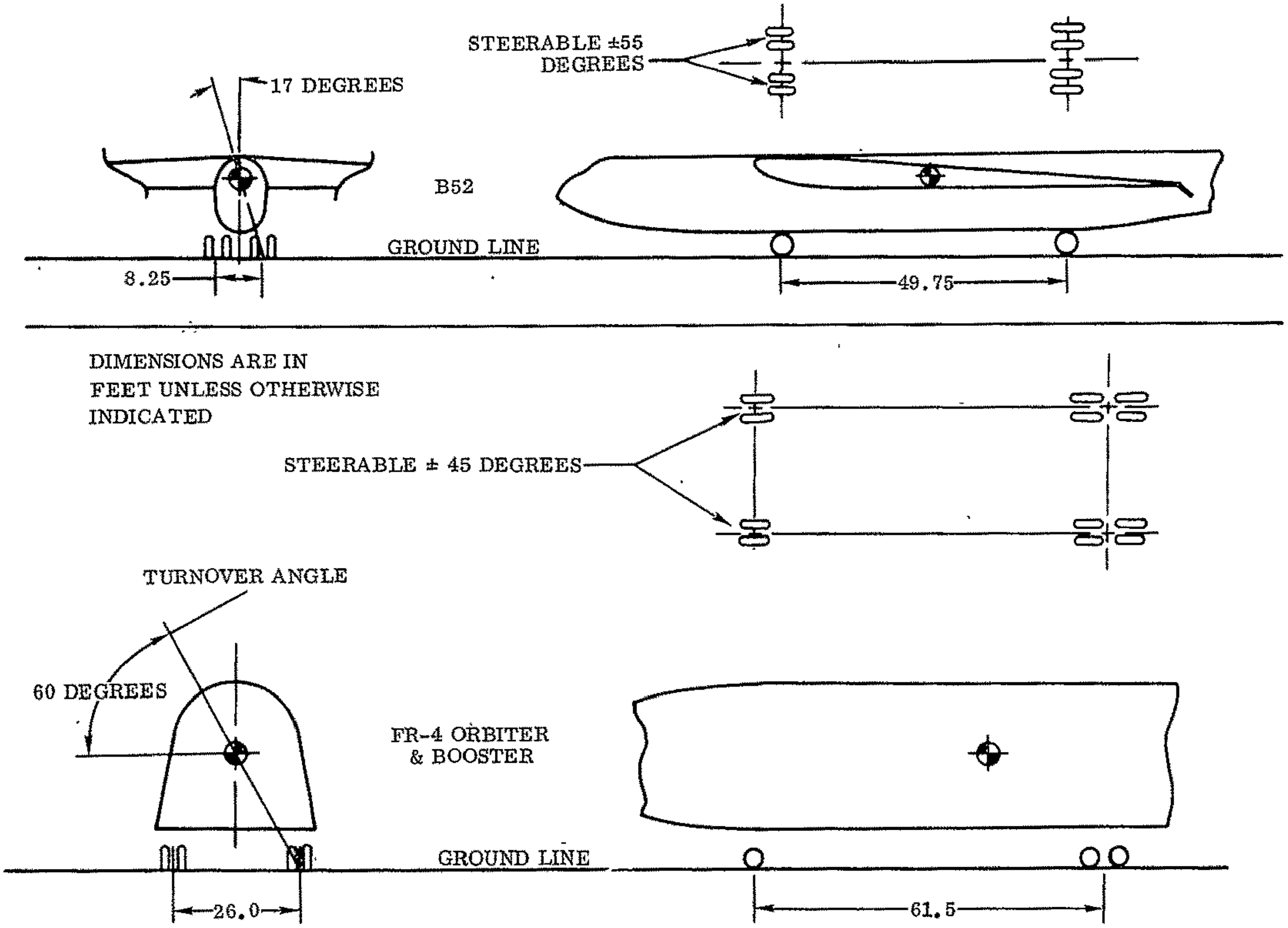

STEERABLE ±55 DEGREES
17 DEGREES
B52
GROUND LINE
8.25
49.75
DIMENSIONS ARE IN FEET UNLESS OTHERWISE INDICATED
STEERABLE ± 45 DEGREES
TURNOVER ANGLE
60 DEGREES
FR-4 ORBITER & BOOSTER
GROUND LINE
26.0
61.5

Figure 9-5. Landing Gear Arrangement

SECTION 10

STAGE SEPARATION

This section documents the work accomplished to date on the separation subsystems for space shuttle concepts with parallel-mounted elements.

The separation system is the outgrowth of several studies to configure separation systems for clusters of aerodynamic lifting vehicles. The current separation concept (the use of an aft hinge linkage system) was selected for an early configuration and is detailed in Reference 10-1. An alternate scheme (parallel free-body separation using booster vehicle's inboard engines) was additionally investigated but dropped in favor of the aft hinge linkage system; this alternate scheme is detailed in Reference 10-2.

A design tradeoff study was then initiated to investigate possible aft-hinge linkage systems. A simple hinge was not feasible since the over-hang aft of the aft-structure hinge attach point on the booster vehicles created collision problems even after a few degrees booster rotation. (Under this concept rotation is necessary to produce the required lift to "fly" the boosters away from the orbiter.) Remaining were a modified simple hinge (one with a single rotating link and a controlled-displacement folded link) and an adaptation of a conventional four-bar linkage system. This design study, detailed in Reference 10-3, concluded that the modified aft-hinge results in a superior system, which was subsequently selected as the linkage system.

Supersonic and low-hypersonic wind-tunnel tests were then conducted to investigate the interference aerodynamic properties and to demonstrate the separation trajectories under two widely varying Mach conditions. These test results indicated that the aerodynamic interference effects are indeed quite large when the vehicles are in proximity but fall off rapidly to essentially isolated-body aerodynamics for moderate lateral displacements. Separation trajectories were demonstrated for a variety of hinge-release angles (and angular rates) and were particularly graphic in the tunnel tests exhibiting a high-q, low-Mach abort condition since the maximum angular excursion was exhibited in the conventional and Schlieren movie coverage.

The application of this basic separation system to FR-1 incorporated further design refinement in the process:

a. A displacement control member was added to control (restrain) deployment of the lower (folded) link and eliminate deployment impact loads.

b. The displacement control member, used as a actuator, provides the linkage and force for stowage of the separation linkage.

c. The orbiter vehicle's engines are simultaneously throttled to:
   1. Improve the clearance between the boosters and the orbiter engine exhaust plume.
   2. Reduce the required booster release angle.
   3. Reduce the release angular rate.
   4. Reduce the maximum booster angle of attack.

Section 10.1 summarizes the main results from previous studies and describes the analyses leading up to the final FR-1 separation system. Sections 10.2 and 10.3 describe work completed on the FR-3 and FR-4 configurations.

10.1 SUMMARY OF APPLICABLE BACKGROUND STUDIES

In an effort to uncover the most likely abort situations, a brief survey of Atlas launch vehicle failures was undertaken. Atlas was chosen since the boost trajectory and the flight hardware operating during the boost both are similar. Further, Atlas has one of the largest operational flight records (approximately 350 flights of a near consistent configuration) and readily available failure records. It was concluded that situations can occur (mainly associated with an irreversible situation) where an immediate abort would be required. (By immediate abort is meant a situation where it is impossible or too dangerous to delay the interstage separation to achieve a more optimum separation environment.) The abort procedure was formulated as follows:

a. In an immediate abort situation, initiate separation immediately.

b. In an nonimmediate abort situation, take appropriate action (e.g., cut-off of a damaged engine) and proceed to a set of conditions more conducive to separation before initiating the separation sequence.

Since the majority of immediate abort type situations on Atlas occurred midway or beyond in the boost phase, it is reasonable to assume that sufficient altitude and velocity (energy) will have been gained to ensure time for separation, stabilization, venting propellant, wing and engine deployment, etc. before impending ground impact.

The objectives of candidate separation schemes were then formulated as follows. Any separation scheme:

a. Should utilize a simple, reliable separation mechanism, preferably one that has been proven in similar applications.

b. Should be a passive scheme if possible, e.g., not require auxilary thrusters nor the main booster engines.

c. Should be readily extendable to an abort situation with maximum probability of saving of equipment.

d. Should provide an acceptable level of acceleration to the pilots so as not to degrade their piloting function.

e. Should maximize — to the extent practicable — the energy imparted to the orbiter, e.g., minimize time that orbiter is supporting the boosters and minimize unburned booster propellants.

f. Should provide clearance between the boosters and the turbulent region caused by the orbiter exhaust plume in most instances.

g. Should maximize the clearance-time profile (especially under abort conditions to minimize the explosion hazard).

h. Should avoid potential adverse hypersonic aerodynamic properties at large angles of attack by attempting to limit α to 60 degrees or less.

An aft hinge separation of the booster elements from the orbiter element while the orbiter is under thrust was selected as being the most promising scheme consonant with these objectives.

Several of the stated objectives can be met at the onset utilizing an aft hinge system:

a. The aft hinge separation mechanism has been successful for many aerodynamic fairing ejections (e.g., for the Atlas-Centaur, Surveyor, and OAO-A2 missions) and was a scheme in strong contention for Titan-III to separate their strap-on solid motors.

b. The scheme is passive in that the boosters rotate off the hinge due to their offset mass center in conjunction with the accelerating orbiter. For the range of dynamic pressures expected during the ascent phase, no auxiliary thrusters would be required; however, the orbiter must be under thrust.

c. The scheme is readily extendable to an abort situation (as Figure 2, Reference 10-1 amply demonstrates).

d. From the result to date, it is reasonable to conclude that the restrictions imposed on the pilot's environment will never be exceeded.

e. A limiting case of this scheme (a zero-degree hinge angle at release, i.e., an instantaneous release) maximizes the energy imparted to the orbiter to the extent practicable (i.e., as shortly after shutdown as is safe). This maximization comes about since the boosters can be allowed to burn to depletion before the initiation of engine cutoff. Only unburnable residuals and residuals due to errors in the propellant utilization system (both intrastage and interstage) need remain. However, delayed (controlled) release will be necessary to meet the remaining objectives and will incur additional, although small, performance losses.

The remaining objectives to be satisfied are:

f. The staging boosters should clear the turbulent region caused by the orbiter exhaust plume.

g. The scheme should maximize the clearance-time profile (especially under abort conditions to minimize the explosion hazard).

h. To avoid potential adverse hypersonic aerodynamic properties at large angles of attack, attempt to limit α to 60 degrees or less.

These objectives can be met through control of the remaining system variables: the hinge angle at release and the minimum allowable dynamic pressure at separation initiation.

The following separation system parameters were selected for an early FR-1 configuration following a preliminary parameter study:

a. The hinge release angle should be 25 ± 5 degrees.

b. The nominal/minimum dynamic pressure at staging is 50/40 psf.

c. The booster aerodynamic controls should be activated shortly following disengagement (to restrict α to 60 degrees or less).

d. The performance penalty of this separation system over an optimum (zero hinge release angle) system is 32 ± 2 fps orbiter velocity loss.

e. The maximum rigid-body hinge loads were estimated to occur at conditions not analyzed (i.e., near maximum dynamic pressure). The calculation of these hinge loads was deferred.

These separation system parameters provided satisfactory performance over a wide range of nominal and abort conditions. (See Reference 10-1 for full details.)

The parallel free-body separation scheme was briefly analyzed for applicability (details are contained in Reference 10-2). For this scheme, it is the inboard engine that provides the lateral impulse to initiate separation, being gimbaled through the booster's cg so as to produce no rotational moment (which could cause tip or tail collision in short order). Two arrangement stacks are possible: the boosters in a belly-to-belly sandwich about the orbiter (that proposed for investigation) or a similar back-to-back sandwich. The arrangements differ in that the lateral cg offset is located toward the booster's belly; the resultant initial lateral separation acceleration is therefore correspondingly larger for the back-to-back stack.

The following conclusions are reached:

The belly-to-belly stack:

a. Provides unacceptable initial separation acceleration and unacceptable clearance-time envelopes.

b. Requires unattractive alternate techniques to the fixed-angle thrusting booster engine (e.g., activating booster engine control in the proximity of the orbiter and/or forward thrusters to attain early positive angles of attack with an aft hinge).

The back-to-back stack:

a. Provides unacceptable separation acceleration at aerodynamic trim conditions and the probability of reconnection (collision).

b. Requires unattractive alternate techniques to the fixed-angle thrusting booster engine (e.g., activating booster and/or aerodynamic control in the proximity of the orbiter to maintain zero or negative angles of attack).

Both schemes:

a. Require an elaborate throttling sequence before separation can be initiated.

b. Require use of the inboard booster engine to achieve separation under either nominal or abort separation conditions.

c. Require propellants plus propellant reserves onboard the booster elements that cannot be used to impart energy to orbiter.

In conclusion, the proposed lateral separation scheme had several serious drawbacks not characteristic of the aft-hinge scheme outlined above, in addition to being non-passive.

Reference 10-3 describes two candidate systems for separation of the FR-1 Reusable Launch Vehicle/Spacecraft. The two systems are: 1) the single rotating link (aft-hinge), and 2) the double rotating link. The analysis of these systems was on the basis of the FR-1 IPD vehicle per drawing GDC 69-59886; vehicle gross liftoff weight was 485,000 pounds each for boosters and orbiter.

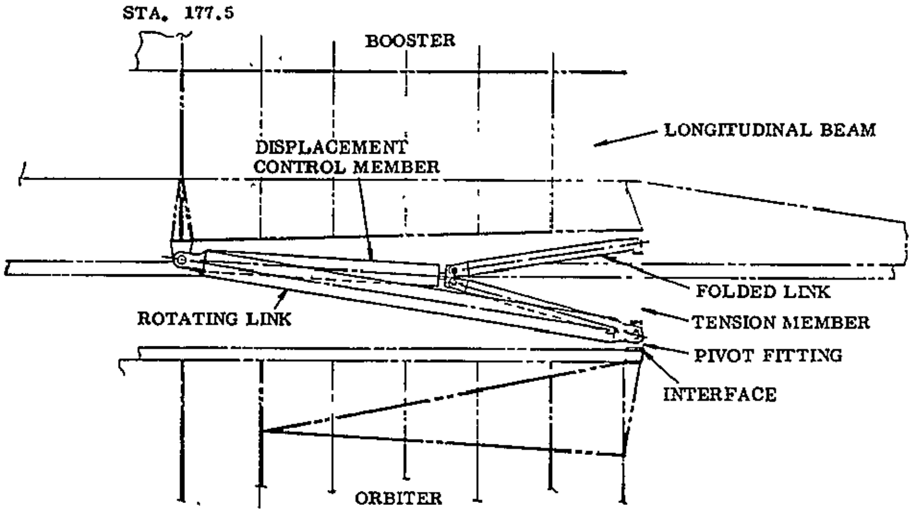
The single rotating link aft-hinge system consists of a pair of rotating links attached with a pair of folded arm/shock absorber links that are attached between each booster vehicle and the orbiter vehicle. The linkage is attached to the orbiter vehicle at the intersection of the rotating link and folded link and is attached to the booster vehicle at the other two ends. A pair of these links works in parallel, thus providing lateral stability by the cross-member between the pair of rotating links.

The double rotating link system consists of a pair of two rotating links attached between the orbiter and booster vehicle, which act as a four-bar linkage. This separation system provides positive displacement and rotation control of the booster vehicles during the separation sequence until release occurs.

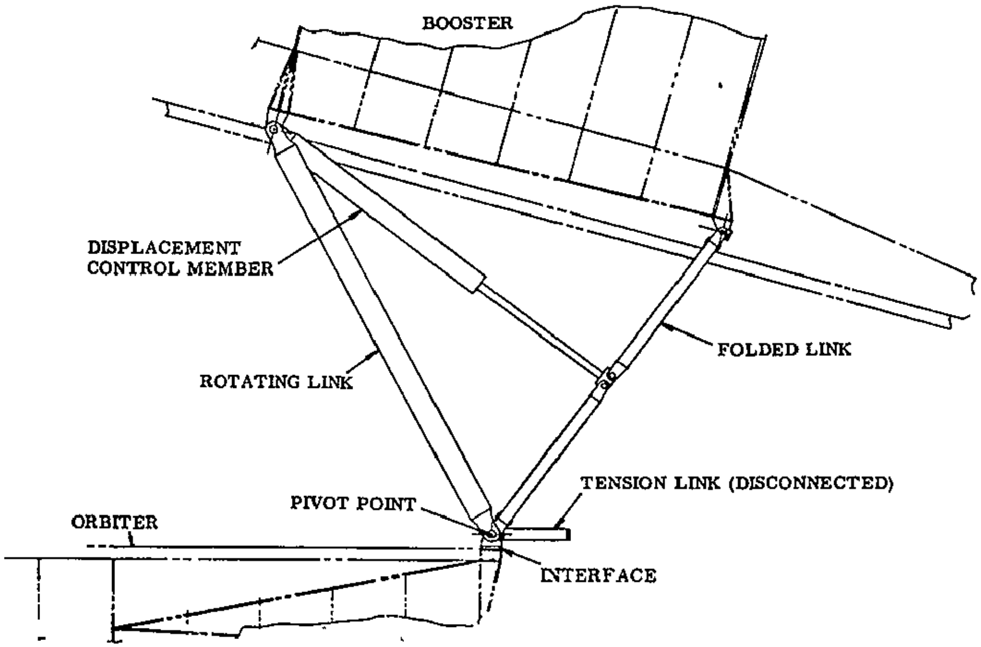
The objective of the analysis (presented in Reference 10-3) was to find a linkage configuration that will most satisfactorily separate the vehicles. Results indicate that the single rotating link aft-hinge separation system is superior and is a feasible system. The pair of linkages at the aft end of each booster vehicle act as a translating hinge about which the boosters can rotate on the orbiter. At staging, with the booster propellants depleted and engine thrust terminated (orbiter engines continuously thrust), the forward interconnects are released. The booster vehicles are then free to rotate about the aft linkage pivot points, by the moments from the booster inertia and the aerodynamic lift and drag forces. The moments from inertia forces are approximately 10 times larger than the aerodynamic moments at the nominal separation conditions, thus assuring positive pitch up of the boosters. The aft linkage translates the booster vehicles outward from the orbiter to provide adequate aft end clearance during rotation. The linkage system is configured to utilize existing forces (no actuator or active controls required) to accomplish the translation. After the booster vehicles have rotated 15 to 20 degrees, the linkages are disconnected, freeing the vehicles of each other. At 15 to 20 degrees of rotation (and, correspondingly, angle of attack) the aerodynamic lift forces are large enough to fly the booster vehicles away from the orbiter and its exhaust plume. The linkages are retained on the booster vehicles and retracted into the heat shield.

An analysis was then undertaken to further definitize the FR-1 orbiter plume structure at the nominal separation conditions. A secondary objective was to obtain the environment at various points in the resulting shock layer to assess the consequences of dipping the booster aft-structure into the plume. This analysis (presented in Reference 10-4) gave results that agreed well with a "plume boundary" plot obtained from Pratt and Whitney although for slightly different conditions. Plume structure to 40 exit plane radii and the Mach number, pressure, and temperature for various points in the shock layer were obtained. These results (contained in Reference 10-4) provided the basis for the FR-1 separation analysis.

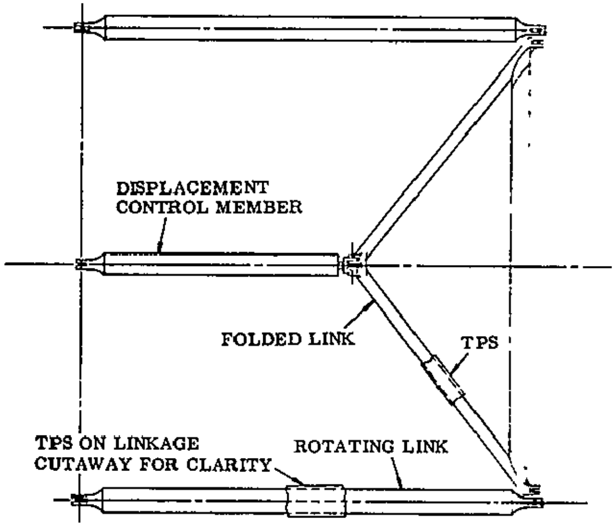
10.1.1 FR-1 SEPARATION LINKAGE DESCRIPTION. The stage separation linkage (Figures 10-1 and 10-2) consists of rotating links, folded links, and a displacement control member. The function of the linkage is to prevent interference between vehicles as the booster rotates about the pivot points during the stage separation sequence. The booster inertia and the aerodynamic forces cause the boosters to rotate outward while the linkage unfolds, thus providing clearance between the aft ends of the vehicles. The displacement control member controls the rate at which the folded link unfolds to extend the linkage. The forces in the linkage are such that the displacement control member (which is a hydraulic snubber) is always in tension during the rotation phase.


STA. 177.5
BOOSTER
LONGITUDINAL BEAM
DISPLACEMENT CONTROL MEMBER
FOLDED LINK
ROTATING LINK
TENSION MEMBER
PIVOT FITTING
INTERFACE
ORBITER
STA. 179.5

SIDE VIEW OF LINKAGE (BEFORE STAGING)


BOOSTER
DISPLACEMENT CONTROL MEMBER
FOLDED LINK
ROTATING LINK
TENSION LINK (DISCONNECTED)
PIVOT POINT
ORBITER
INTERFACE

SIDE VIEW OF LINKAGE IN EXTENDED POSITION (BOOSTER ROTATED)


DISPLACEMENT CONTROL MEMBER
FOLDED LINK
TPS
TPS ON LINKAGE CUTAWAY FOR CLARITY
ROTATING LINK

PLAN VIEW OF LINKAGE

Figure 10-1. Separation Linkage Details

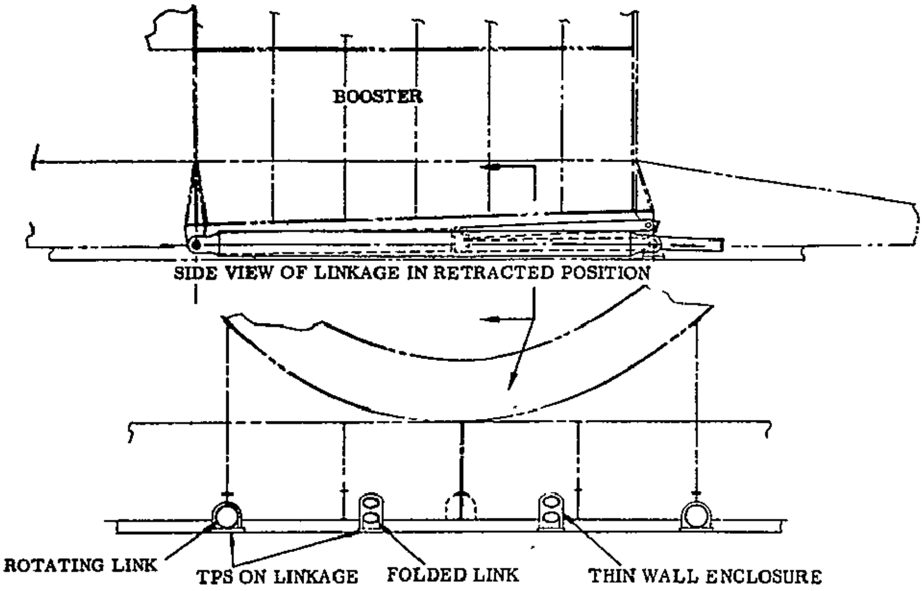


Figure 10-2. Separation Linkage in Retracted Position

The aft attachment points are located at the aft frame of the orbiter thrust structure at Station 190 feet. These points form the hinge line for the booster to rotate away from the orbiter during the separation sequence. The two aft points have their interface at the outer contour line of the orbiter. Two tension members and the stage separation linkage reach out from the booster to the attachment points on the orbiter.

Before the staging sequence is initiated, the two tension members carry the z direction forces and the stage separation linkage carries the lateral (y direction) forces. At initiation of staging the two tension links are separated simultaneously with the forward attachment point. The stage separation linkage, including pivot fittings, are separated at the interface line when the boosters have reached the desired rotation angle.

After separation is complete the linkage system is retracted by the displacement control member (now used as an actuator) into the booster vehicle heat shield (Figure 10-2). The linkage members have thermal protection on one side so that when retracted the booster lower surface is flush and thermally protected. The linkage is isolated from the aft end structure by a thin gage heat-resistant metal box to prevent hot gases from penetrating the vehicle if the seals around the retracted linkage should leak.

10.1.2 PRELIMINARY SEPARATION TRAJECTORY TRADEOFFS FOR FR-1. The separation trajectories were computed assuming that the orbiter is not perturbed from its pre-separation conditions until the boosters are released. Each booster is simulated in 3 degrees of rigid-body freedom: longitudinal (x) and vertical (z) translation and pitch (θ) rotation. Detailed dynamic studies in 6 degrees of freedom for all three bodies demonstrate that ignoring the details of booster-orbiter inertial interaction is an adequate assumption for preliminary analyses and is considerably more economical. Interference aerodynamics as obtained from recent wind tunnel testing were, however, included. Link load histories and the post-separation booster capture maneuver were by-products of the computer trajectory simulation.

The linkage system was simulated as shown in Figure 10-1. The displacement control memeber was programmed to provide zero outward velocity of the folded link at full link extension to obviate link impact loading. In addition, a smooth transition from the initial position to full extension was provided by selecting a cos (ωt) function for the acceleration of the span (L1) between the booster and orbiter folded link attach points.* This gave a minimum-time smooth extension solution for the folded link but had the disadvantage of a symmetric span acceleration profile (even though the displacement control member has very small leverage in its initial position and very large leverage in its extended position, (See Figure 10-1.) This tended to be a non-optimum acceleration profile but was convenient since it met the boundary conditions automatically. (A more appropriate criterion for future study would be a link load minimum analytic path.)

The linkage configuration as selected was based on previous studies and preliminary considerations for FR-1. The variable link span, L1, and the constant rotating link length, L2 (see Figure 10-1), are based predominately on packaging limitations. It is desirable to locate the pivot point as far aft as possible. The aft structure of the vehicles ends at approximately Station 190 feet. The preliminary analysis indicated that with L2 equal to or less than 152 inches, L1 would have tension forces at all times and thus the displacement control member need be only a damper rather than an actuator, a highly desirable feature. Increasing the length of L2 increases the separation of the booster but puts the displacement control member in compression when at its poorest mechanical advantage. Moving the pivot point location aft of Station 190 feet requires that structure be extended aft of existing thrust structure. Thus, both L2 length and pivot point location are not at optimum values. Since with L2 greater than 150 inches in length penalties occur because the displacement control member must be an actuator, and structural penalties occur with the pivot point located aft of Station 190 where vehicle primary structure ends, these values were selected as being the most optimum consistent with the existing configuration design.

The initial separation trajectories are shown in Figure 10-3 for release angles of 10, 15, and 20 degrees. These are orbiter-relative trajectories with the booster trimmer at zero deflection and the orbiter at 100% thrust. (The orbiter engine plume geometry was obtained from Reference 10-4.) The 20-degree release angle is sufficient to just cause collision of the orbiter engine and the trailing edge of the booster

---

*This span was simulated as a psuedo-link, L1, which tied the link attach points together and was extendable. This says that $d^2 (L1)/dt^2 = k\cos(\omega t)$; k and ω constant.

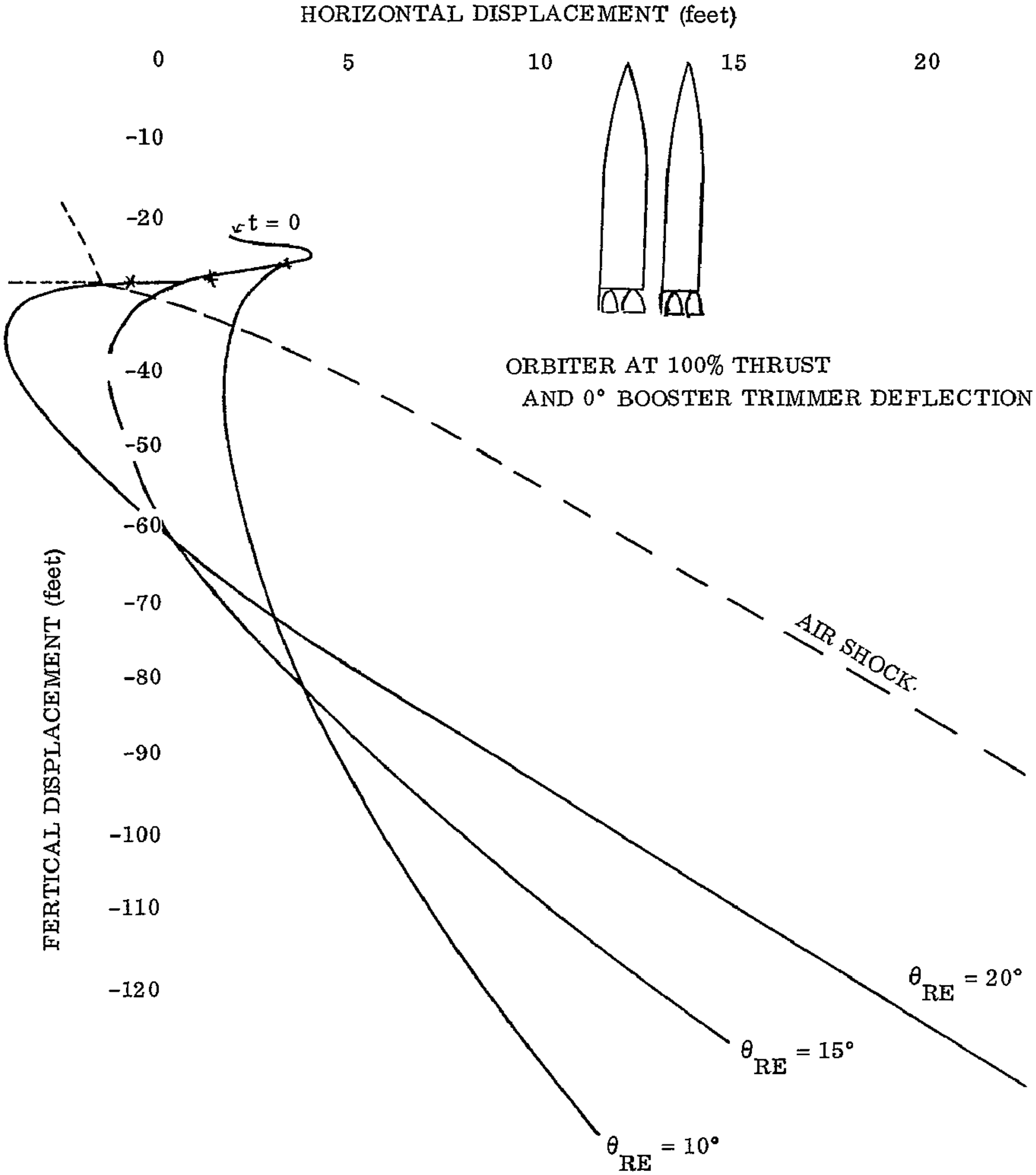


Figure 10-3. Separation Trajectories for Various Hinge-Release Angles

trimmer. The study configuration aerodynamics* do not provide sufficient lift (at the staging q of 50 psf) to clear the orbiter exhaust plume (Figure 10-3) nor sufficient aerodynamic control moment capability to prevent the tumbling boosters from going fully broadside after disconnection. The 9-foot aft extension of the trimmer surface to improve hypersonic trim has seriously degraded clearance at moderate release angles due to the now large overhang aft of the release hinge. Deflections of the trimmer surface to provide corrective moments for the boosters further aggrevate the clearance problem.

A principal problem — that of providing suitable separation clearance with the orbiter engine exhaust plume — can be partially eliminated by throttling the orbiter engines while the boosters are rotating. Figure 10-4 illustrates the improvement attained by throttling the orbiter thrust at 35% per second to 20% minimum. This throttling was initiated simultaneously with detaching the boosters and had reached the 20% minimum by booster release. Although plume clearance is greatly improved (clearance at 5.4 seconds following initiation of separation is indicated on the figure) initial clearance of the trailing edge of the booster trimmer and the orbiter engine bell — especially if this engine is gimballed — is still of major concern. Further improvement by reducing the booster release angle will not provide sufficient lift, and the booster will again drop through the plume.

Investigation of the booster trimmer indicates that it could be gimballed nearly 15 degrees inboard without interference if the booster's engines are in the stowed (bells topside) position. Substantial clearance can be gained in this way at the possible expense of further aggravating the post-separation booster rate of tumble. Figure 10-5 illustrates the improvement (over the preceeding figure) for an intial 10-degree up-deflection of the trimmer. Upon release, the trimmer was repositioned to +20 degrees at +10 deg/sec. As noted on the figure, a light loss in displacement at 5.4 seconds following initiation has occurred; note however that the orbiter's final thrust level has been reduced to 10%. For these conditions there was no appreciable change in booster rate of tumble.

Figure 10-6 illustrates the effect of an addition of a 30,000-pound nose jet on separation clearance. In the vicinity of the orbiter the ensuing reduction in rotational rate has an immediate effect in increasing clearance. As expected, the addition of the nose jet to offset the tumble rate has reduced the separation clearance (i.e., it thrusts towards the orbiter**) as can be noted by comparing Figures 10-5 and 10-6. Table 10-1 presents the calculations for the nose jet parameters. The weight is based on

---

*These aerodynamics are based on extensive wind-tunnel testing but represent a significant departure from previous results based on analytic aerodynamics, e.g., see earlier separation trajectories presented in Reference 10-1. They gave rise to an interim version of the FR-1 design that had a long afterbody trimmer to improve the hypersonic trim point. It is this configuration that was used as the study configuration.

**A tail jet would be better in this regard but has the disadvantage of impinging on the orbiter.

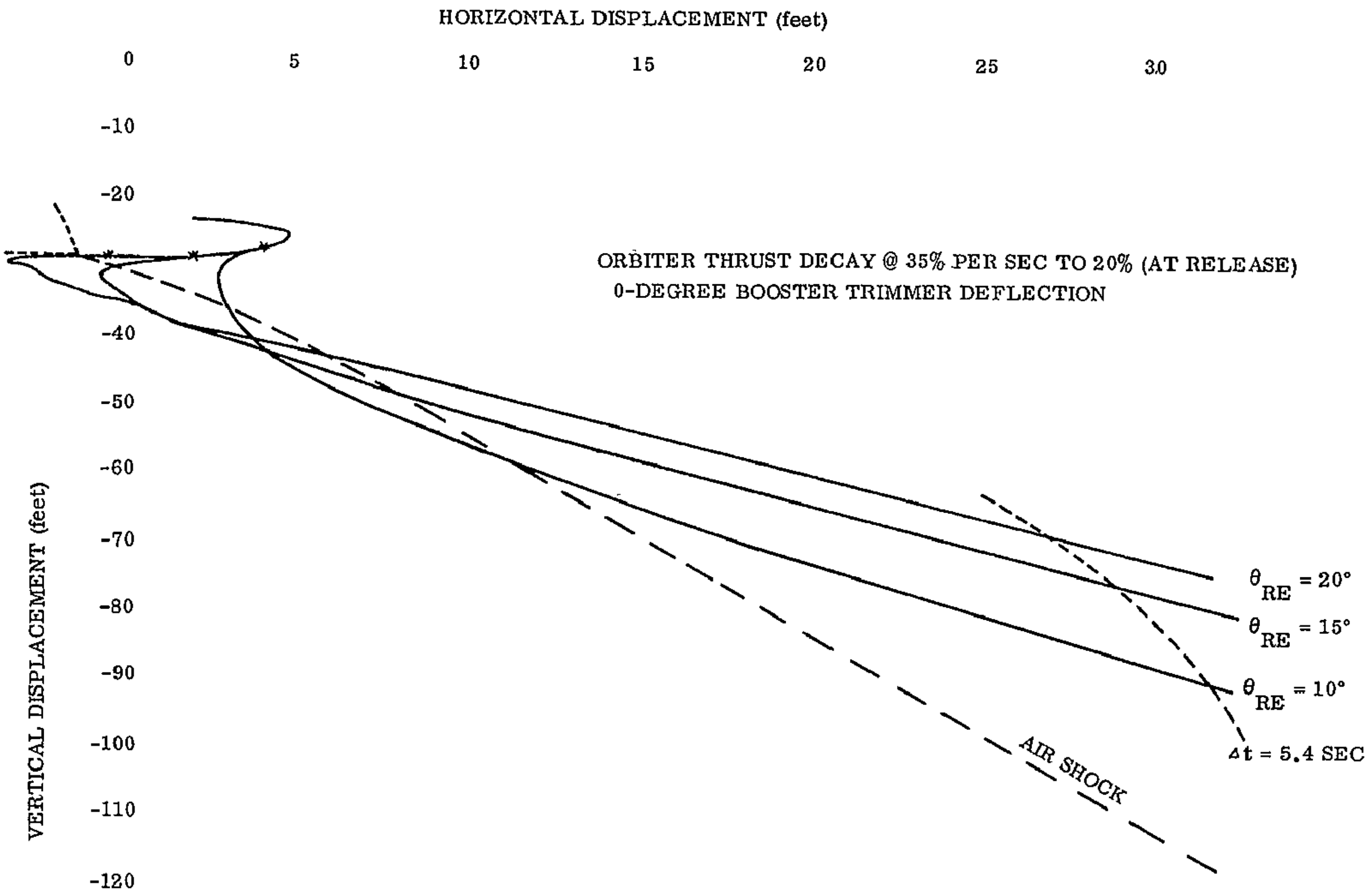


Figure 10-4. Separation Trajectories for Various Hinge-Release Angles with Thrust Decay

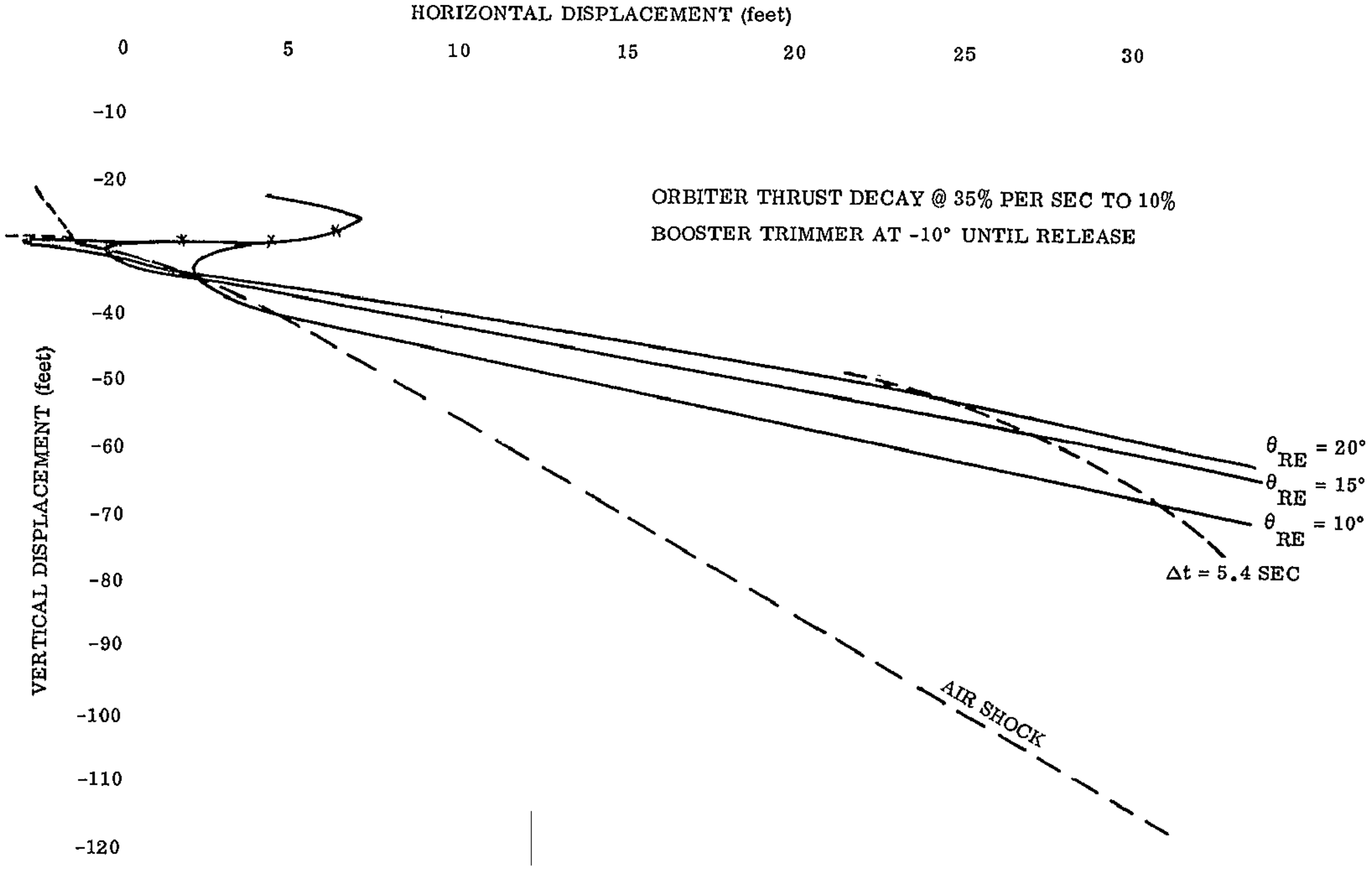


Figure 10-5. Separation Trajectories for Various Release Angles with Trimmer Stowed

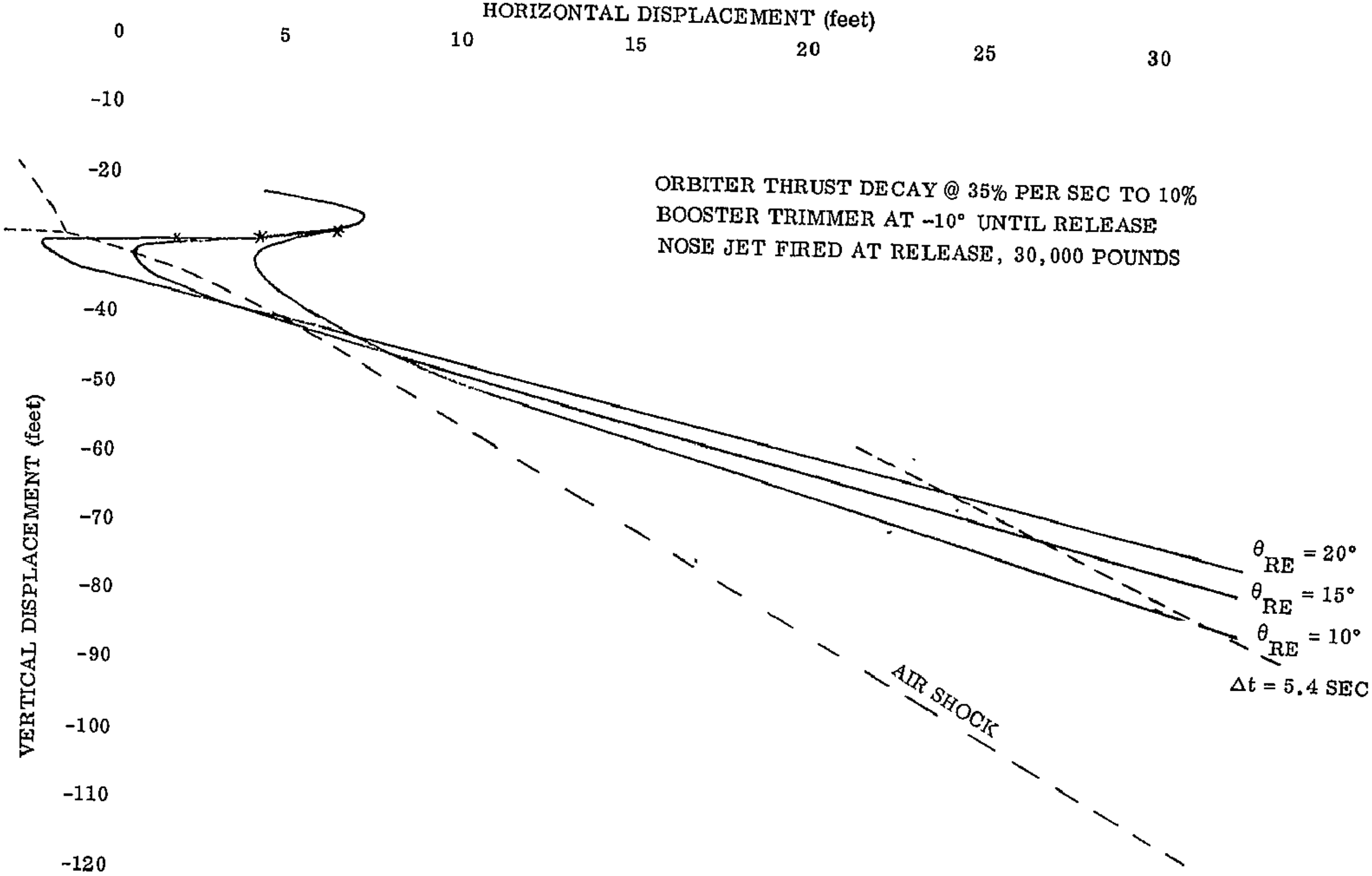


Figure 10-6. Separation Trajectories for Various Release Angles with Nose Jet

Table 10-1. Nose Jet Parameters

θ_{RE} (deg)	$\dot{\theta}_{RE}$ (deg/sec)	θ_{MAX} (deg)	Δt_{JET} (sec)	W_{JET} (lb)
20	15.11	60	4.6	590
15	12.45	45	4.0	510
10	9.72	29	3.4	435

an ISP of 270 seconds and a mass fraction of 0.87. Since the boosters trim at approximately 48 degrees, this jet is nearly ideal for the 15-degree release angle (θ_{max} = 45 degrees). It is noted that the weight penalty is minor.

An examination of the performance of the booster on the orbiter backside, Figure 10-7, indicates that all release angles examined will intersect the orbiter plume shock. This arises predominately from the orbiter upper engine being over one foot outboard of the skinline (VRS over one foot inboard, Figure 10-6). In addition, the interference aerodynamics (as obtained from the wind tunnel tests) are not as effective on the orbiter backside. These aerodynamic contributions are particularly evident from comparing the clearance at 5.4 seconds on Figures 10-6 and 10-7. The latter figure points out that, in order to insure clearance with the orbiter engine bell, a 10-degree release angle must be selected; nevertheless, the topside booster will intersect the plume-air shock boundary with the trailing portion of the trimmer surface.

It is evident from the foregoing that the booster trimmer will intersect the plume-air shock boundary. This is particularly true for either booster if the orbiter engines are allowed to gimbal. With a 5-degree out-board deflection of the orbiter engine, approximately 2.5 feet of clearance are lost and the air-shock is rotated up 5 degrees.

Using the results presented in Reference 10-4, it was possible to make a rough approximation as to the effect of the plume-air shock on the booster trimmer. Calculations* indicated that between the jet plume boundary and the air shock, a maximum of 2.5 psi might be imposed on the trimmer. If the trimmer dipped into the jet plume itself, up to 4.0 psi might be imposed. It was conceded that the trimmer can be designed for these loads. Since the immersion duration is small, the temperature rise on the trimmer surface will be small and can be ignored. (Figure 3c, Reference 10-4, gives the ambient temperatures in the shock layer.) Thus, it is assumed that it is permissable to dip the trimmer into the plume; indeed, the resulting corrective moment could approach that supplied by the nose jet.

---

*These calculations used a pressure coefficient $C_p = 2 \sin^2 (\alpha_T)$ where α_T is the angle of attack of the trimmer with the various shock layers.

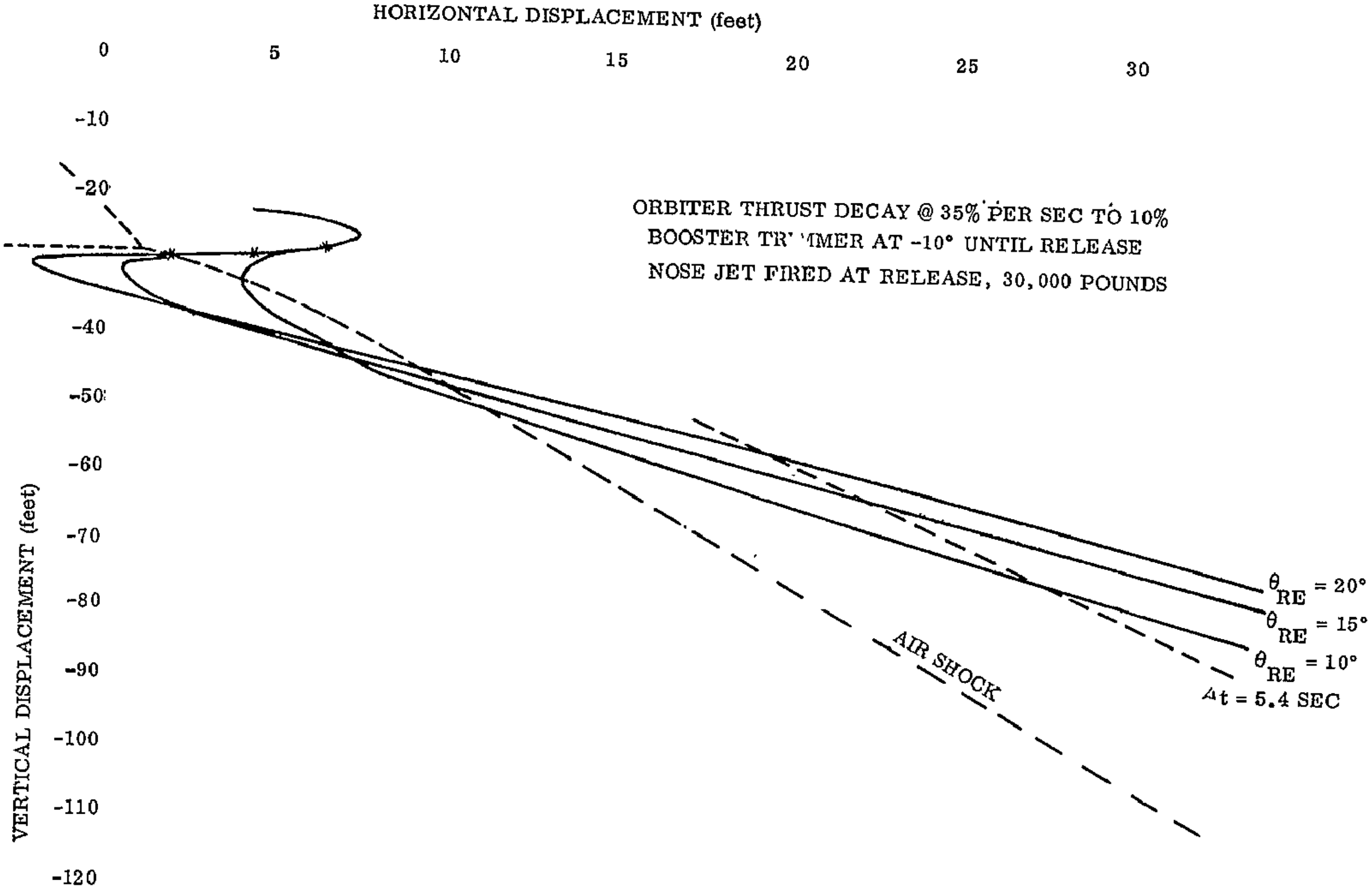


Figure 10-7. Separation Trajectories for Various Release Angles, Topside Booster

Figure 10-8 presents the link loads during the rotation prior to disengagement. Shown on the figure are the booster hinge angles (θ) corresponding to the release angles under investigation. Note that the link displacement control member (T3) is initially in compression for the linkage system under investigation. This indicates that, for the initial 0.35 second, the displacement control member is acting like an actuator and then like a damper. These loads are well within the design realm.

10.1.3 FR-1 STUDY SUMMARY. As envisioned, the booster elements would thrust to propellant depletion subject only to maintaining the dynamic pressure at staging of not less than 50 pounds per square foot. (This minimum dynamic pressure is required to provide sufficient lift to enable the boosters to "fly" off the orbiter without resort to auxiliary lateral thrusters or the main thrust engines.) Upon sensing the combined depletion signal (both interstage and intrastate), a thrust termination signal is sent to both boosters and the orbiter begins a pre-programmed throttle mode. Fast booster thrust termination is possible since the booster engine system propellant head is quite low owing to being near propellant depletion.

Separation is initiated when booster thrust has reached a "commit" level by releasing the forward attach points and the two aft tension links (tied into the separation linkage). The boosters are then free to rotate about the aft hinge and will do so under the combined aerodynamic and inertial reaction load provided by the still thrusting orbiter element. Nearly symmetric booster rotations are obtained due to the nearly identical mass properties of the depleted booster elements (the dominate force term).

When the desired booster rotation angles are obtained, the linkage system is separated flush with the orbiter's skin-line by redundant explosive nuts. The displacement control member is then used as an actuator to retract the linkage into the vehicle heat shield and provide additional clearance.

The boosters, upon disengagement, initiate a capture maneuver (with the aid of a fixed-impulse nose jet) and trim to high lift. The orbiter element, having been programmed to minimum (idle) thrust, remains at idle thrust to enhance clearance between the vertically accelerating orbiter engine plumes and the now decelerating but lifting boosters.

When sufficient clearance has been established, the orbiter is brought back to the desired throttle ratio and continues on its mission. The upper booster initiates a 180-degree roll to reorient its left vector to minimize its apogee altitude. Both boosters hold high lift through apogee and until sufficient dynamic pressure again becomes available to initiate a 180-degree roll for reentry.

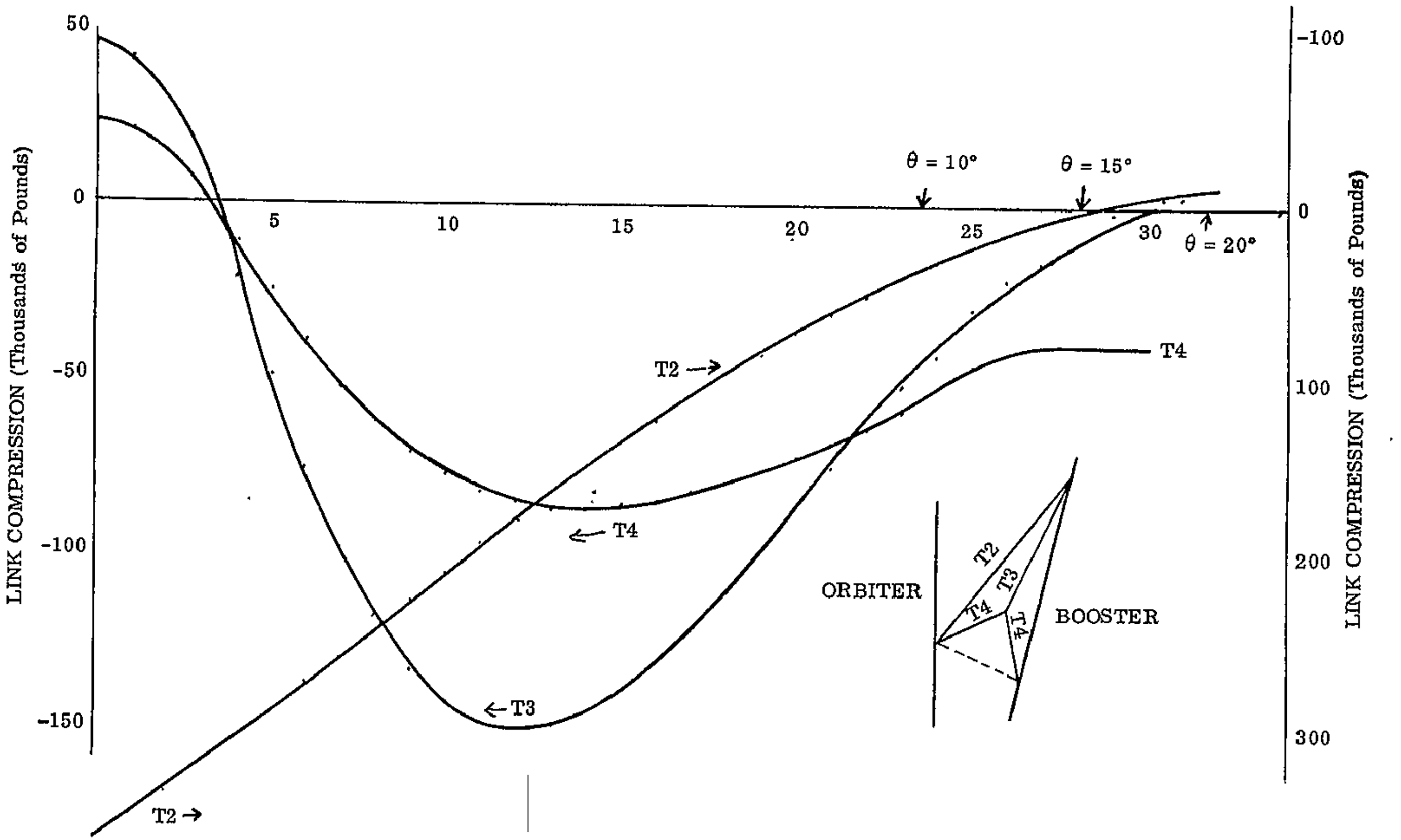


Figure 10-8. Link Load History

The major concern of the tradeoff study was to provide clearance between the booster afterbody and the orbiter main engines (initially) and engine exhaust plume shock region (subsequently). It was demonstrated that, even with the very large afterbody extention (being proposed at that time), it was possible to provide suitable clearance with the engine; however, the afterbody would dip into the air shock boundary layer generated by the orbiter engine plumes. Figures 10-6 and 10-7 show typical orbiter-relative trajectories.

10.1.4 FR-1 STUDY CONCLUSIONS. Although a detailed parameter study was not undertaken, it is possible to draw several conclusions from this study regarding the FR-1 launch configuration with the T-18 boosters and orbiter.

a. It would be desirable to place the T-18 configuration's engines along their pitch axis. This would yield on additional 7.5 feet of clearance between the orbiter engine plumes and the boosters and eliminate dipping the booster's trimmer into the plume.

b. It would be desirable to fly the FR-1 stack horizontally rather than vertically. This would mean that each booster upon disengagement would need to perform a 90-degree roll maneuver (to a belly-up attitude) to reduce their apogee altitude. Presently the topside booster must reorient 180 degrees and the bottom side booster not at all.

c. It would be desirable to alter the T-18 configuration geometry to provide better hypersonic basic body stability without the requirement for a long after-body extension. This would allow the separation hinge to be located more favorably (i.e., closer to the booster aft end) and substantially reduce the clearance lost through rotation.

d. It is desirable to modify the link extension acceleration profile to minimize the link displacement control member load.

Despite the disadvantage of having the booster trimmer dip into the plume shock, the parameters selected give reasonable performance. The link loads incurred during the nominal separation are commensurate with those obtained in previous studies. With minor adjustments, the parameters selected, although not optimum will suffice.

The final T-18 vehicle which does not have a 9-foot lower surface extension would have improved separation characteristics.

10.2 FR-3 STAGE SEPARATION

The FR-3 separation system was selected from the initial concept definitions presented in Volume III, Section 2.5. The longitudinal differential-drag concept was selected as being the simplest and most reliable. Initially a fully passive system, the addition of a nose jet on the booster element, was deemed necessary in view of the weak aerodynamic environment at the nominal staging point. Preliminary calculations using analytically obtained aerodynamics indicated that the differential acceleration (less than 2 ft/sec/sec) was not sufficient to obtain the desired separation velocity at disengagement.

10.2.1 FR-3 STAGE SEPARATION SYSTEM DESCRIPTION. As envisioned, the booster elements would thrust to propellant depletion and initiate engine cutoff. Fast booster thrust termination is possible since the booster engine system propellant head is quite low owing to being at propellant depletion.

Separation is initiated when the booster thrust has reached a "commit" level by releasing the forward and aft attach points and firing a solid-propellant jet located in the booster nose. The booster is then free to slide aft along the orbiter skinline by means of a pair of booster-mounted rails and orbiter-mounted slides.

As the booster moves aft under the combination of aerodynamic and jet induced drag, the nose jet plume will begin impinging on the orbiter thermal protected lower surface.* This additional impingement load will augment the interference aerodynamic loads in providing lateral-rotational clearance as the last slide leaves the rail. The nose jet thrust is terminated just prior to disengagement such that the separation velocity at disengagement (circa 40 fps) is sufficient to provide enough clearance to prevent post-disengagement collision and provide for sufficient clearance at orbiter engine start.

The booster, upon disengagement, initiates a reorientation maneuver and trims to high lift (belly up) in an attempt to minimize its apogee altitude. The orbiter stabilizes aerodynamically (or with its reaction control system) and begins its main engine start sequence.

10.2.2 FR-3 PARAMETER SELECTION. Table 10-2 was constructed utilizing the computed (but minimal) aerodynamic drag differential and the desired performance. The low-contamination jet I_{sp} was 220 seconds, and the installation mass fraction was 0.80.

---

*The adverse effects of this impingement can be mitigated by selecting a low-contamination solid propellant and sacrificing I_{sp}

Table 10-2. Selected Parameters, FR-3

Item	Value
Rail Length (ft)	100
Disengagement Elapsed Time (sec)	5
Separation Differential Acceleration (g)	0.25
Disengagement Velocity (fps)	40
Nose-to-Tail Elapsed Time (sec)	7.5
Jet Thrust (lb)	150,000
Jet Firing Time (sec)	5
Jet Propellant Weight (lb)	3400
Jet Installation (lb)	4250

10.2.3 FR-3 SEPARATION STUDY CONCLUSIONS. The parameters presented in Table 10-2 give adequate performance at the nominal separation point. Abort separation at much higher dynamic pressures was not examined.

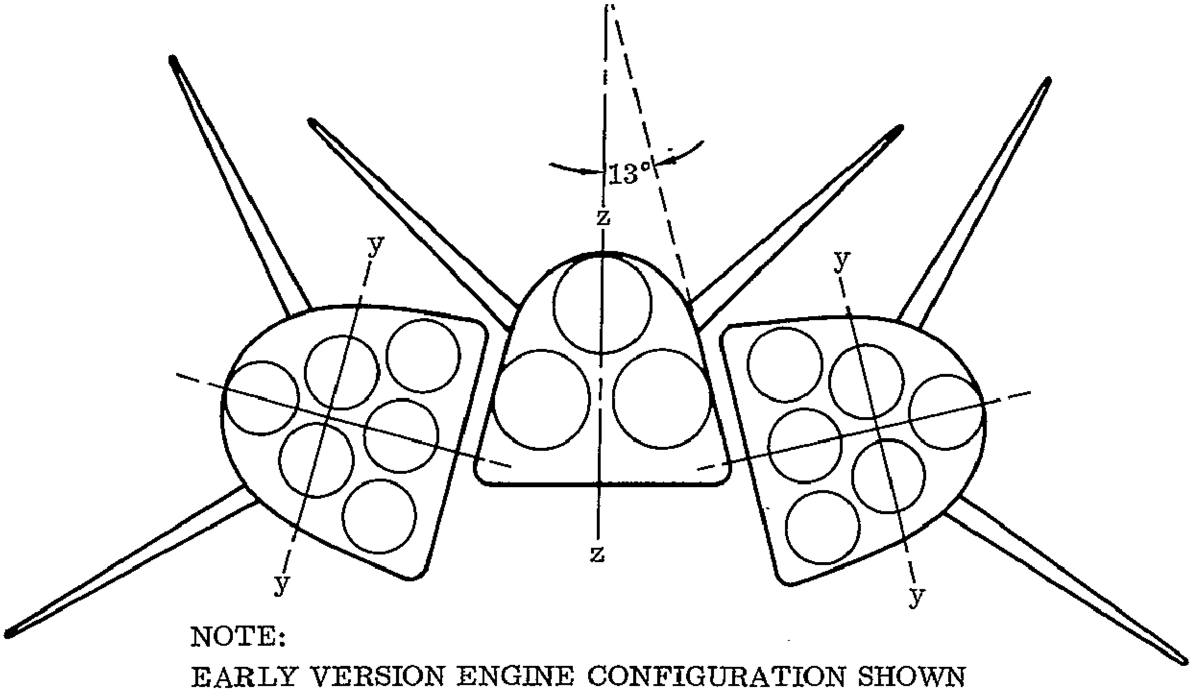
10.3 FR-4 STAGE SEPARATION

Although not specifically analyzed, the FR-4 launch configuration was examined to determine the applicability of the FR-1 aft-hinge staging separation system described in Section 10.1. The lack of symmetry in the FR-4 vehicle* cluster will definitely produce degraded stage separation system performance when compared with FR-1. Although stage separation at the nominal staging point appears adequate, abort separation under much higher aerodynamic pressures appears questionable.

10.3.1 PERTINENT CONFIGURATION DIFFERENCES BETWEEN FR-4 AND FR-1. As configured, the FR-4 vehicle cluster consists of the two boosters nestled along the orbiter sides as close as the orbiter tail will allow (Figure 10-9). The booster y-y axis is aligned 12 degrees to the orbiter z-z axis. The aft-hinge separation system would attach to the orbiter in approximately the same manner as FR-1. This places the attach points approximately 114 feet aft and 13 feet below the orbiter mass center.

---

*This lack of symmetry was dictated in response to a customer requirement for complete payload accessibility up to within seconds of launch.



NOTE:
EARLY VERSION ENGINE CONFIGURATION SHOWN

Figure 10-9. FR-4 Launch Configuration

Presuming an aft-hinge separation system performance similar to FR-1 (cf. Section 10-1), the lack of configuration symmetry can be readily assessed. Assumming the configuration is in trim prior to the initiation of separation (boosters at zero thrust, orbiter at partial thrust), the following effects can be anticipated as aft-hinge rotation is accomplished.

a. The lift and drag components will increase the aft-hinge rotation, i.e., with booster angle of attack.

b. The increased drag component will produce additive axial components at 13 feet below the orbiter center of mass and a nose-down moment.

c. The increased lift component along the orbiter y-y axis will cancel. However, the increased lift component times the sine (12 degrees) is additive and acts at 114 feet aft of the orbiter center of mass and produces an additional nose-down moment.

d. The resultant nose-down rotation will be only partially offset by the orbiter control engines. Additional moments about the booster z-z axis (above those anticipated for the FR-1 configuration) must be absorbed by the linkage system. These moments operating over rather short moment arms are likely to produce large linkage forces.

e. Positive angles of attack result in angles of sideslip on the boosters and could result in the booster colliding with the orbiters tail and thus must be avoided.

f. The downwash from the orbiter tail is likely to yaw and roll the boosters as they depart. (This effect is nearly cancelled in the FR-1 arrangement but results in different trajectories for the two boosters).

g. Although separation clearance is enhanced by throttling the orbiter engines during release, the resulting reduction in orbiter control will aggrevate the nose down rotation.

10.3.2 FR-4 SEPARATION STUDY CONCLUSIONS. Preliminary calculations for the FR-4 configuration indicate that the resulting performance at the nominal separation point is probably adequate; however, much higher linkage loads are likely. Since every effect enumerated here is aggrevated by higher aerodynamic pressure, abort separation under much higher aerodynamic pressures appears questionable.

10.4 REFERENCES

10-1 Preliminary Analysis of a Passive Scheme to Effect Triamese Stage Separation, Memo AD-69-08, Convair division of General Dynamics, 31 January 1969.

10-2 Preliminary Analysis of the Lateral Free-Body Separation Scheme for Triamese, Memo AD-69-09, Convair division of General Dynamics, 7 February 1969.

10-3 Triamese Stage Separation System Linkage Tradeoff Study, Memo RSTS-69-55, Convair division of General Dynamics, 21 August 1969.

10-4 Final Report, Space Transportation Systems (STS) Study, Vol II, Appendix 4, "Vehicle Plume Calculations," Convair division of General Dynamics, November 1969.

SECTION 11

MASS PROPERTIES

FR-3 and FR-4 mass properties in this section are in accord with the Spacecraft Design Data Statements that appear at the beginning of Paragraphs 11.1 and 11.2 respectively. Dimensions and areas in these statements differ from data and drawings shown elsewhere in this report because they represent a last iteration to obtain properly sized vehicles consistent with final weight calculations and propulsion performance data.

Ten percent contingency on dry weight was specified for this contract, and is shown as such elsewhere in this report. In this section, however, the term "growth" has been used for the ten percent weight factor, as it is Convair's understanding that this allowance is not for the purpose of making up for low weight estimates by contractors. To account for weight uncertainties for which the contractor would be responsible, additional weight has been added for both FR-3 and FR-4 under the heading of "contingency." These applications of the words "growth" and "contingency" are in accordance with the definition given in MIL-M-38310A mass properties standard. The contingency added to FR-3 and FR-4 weight is thus included in the weight to which the ten percent growth factor is applied. The amount of contingency added is not a set percentage, but varies with the degree of uncertainty involved and other factors.

11.1 FR-3 CONFIGURATION

The Spacecraft Design Data Statement appears on the first two pages of the following compilation of mass property data for the FR-3 configuration. The Spacecraft Sequency Mass Property Statement and the Spacecraft Summary Weight Statement are on the next two pages. These are followed by the Spacecraft Detail Weight Statements for the FR-3 configuration.

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-3 ORBITER (15' x 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces	Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)	–		–
Surface Area of Above Volume (ft^2)	–		–
Gross Area (ft^2)	1336		1397
Span (ft)	146.9		66.2
Mean Aerodynamic Chord Length (ft)	12.2		23.0
Theoretical Root Chord Length (in.)	162		338
Theoretical Root Chord Max. Thickness (in.)	34.1		40.5
Theoretical Tip Chord Length (in.)	129.0		203.0
Theoretical Tip Chord Max. Thickness (in.)	23.2		20.3

2. Body Structure	Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)		10638	89060	
Pressurized Volume (ft^3)	350			22665
Surface Area, Wetted, (ft^2)			14900	
Maximum Length (ft)			179.2	
Maximum Depth (ft)			26.4	
Maximum Width (ft)			30.9	
Total Wetted Outer Mold Line Volume (ft^3)			89060	
Total Wetted Outer Mold Line Surface Area (ft^2)			14900	

3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)			580	2890	304
Surface Area Delta					
Within 1.0 & 2.0 (ft^2)			2800	14900	3657
Windward Unit Wt. (lb/ft^2)			2.43	2.45	
Leeward Unit Wt. (lb/ft^2)			1.90	1.88	

4. Launch Recovery and Docking	Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)	230000	96000	
Length, Oleo Extended (in.)*	180	180	
Oleo Travel (in.)**	24	24	
Structural		Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)		287000	1.4
Landing		287000	1.4
Limit Landing Sinking Speed (ft/sec)			12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)			20
Airframe Weight (as Defined in AN-W-11) (lb)			

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	3	3
Max SL Static Thrust per Engine (lb)	400000	21000
Total Propellant Tankage Volume (ft^3)	22665	–
Total JP Fuel Tankage Volume (ft^3)	–	63

6. Through 16.	
Maximum Electrical Power (KVA)	23
Pressurized Surface Area (ft^2)	5507
Maximum Mission Duration (days)	7
No. of Crew	2
17. Through 21.	
Maximum Number of Personnel Including Crew	2
Maximum Cargo (lb)	50000
Maximum Cargo Density (lb/ft^3)	4.7
22. Through 27.	
Capacity Propellant Weight (lb)	629558
Capacity JP Fuel Weight (lb)	2868
GENERAL/MISSION DESIGN	
Maximum Design Gross Weight (lb)	926690
Maximum Boost Load Factor 4 g at 926690 G. W.	
Maximum Boost Dynamic Pressure (lb/ft^2)	670.2
Main Engine Specific Impulse (lb-sec/lb) (Vacuum)	459.0
Delta Velocity Available (ft/sec) Ideal	29518
Entry Velocity (ft/sec) (ALT = 400000 FT $\gamma = -1.0°$)	25970
Entry Weight (lb)	289655
Ballistic Coefficient, W/C_DA (lb/ft^2)	–
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 58.6	
Body L. E. = 46.3 UPPER Surf = 0.35 Lower Surf = 11.3	
Factors of Safety (Define) See Volume II, Section 2	
Margins (Define)	
Contingency (Define) - Uncertainty Allowance	8097
Weight Growth (Define) - 10% Dry Structure	21315

*℄ Axle to ℄ Trunnion

**Fully Extended to Fully Collapsed

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-3 BOOSTER (15' x 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces	Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)	–		–
Surface Area of Above Volume (ft^2)	–		–
Gross Area (ft^2)	2901		2283
Span (ft)	201.0		84.3
Mean Aerodynamic Chord Length (ft)	17.3		32.9
Theoretical Root Chord Length (in.)	224		484
Theoretical Root Chord Max. Thickness (in.)	47		58
Theoretical Tip Chord Length (in.)	189		290
Theoretical Tip Chord Max. Thickness (in.)	34		29

2. Body Structure	Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)			235997	
Pressurized Volume (ft^3)	350			129111
Surface Area, Wetted, (ft^2)			26613	
Maximum Length (ft)			210.1	
Maximum Depth (ft)			36.9	
Maximum Width (ft)			41.1	
Total Wetted Outer Mold Line Volume (ft^3)			235997	
Total Wetted Outer Mold Line Surface Area (ft^2)			26613	

3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within 1.0 & 2.0 (ft^3)				3320	1150
Surface Area Delta Within 1.0 & 2.0 (ft^2)				26610	13790
Windward Unit Wt. (lb/ft^2)				1.92	
Leeward Unit Wt. (lb/ft^2)				1.36	

4. Launch Recovery and Docking	Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)	495000	72000	
Length, Oleo Extended (in.)*	200	160	
Oleo Travel (in.)**	24	24	
Structural		Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)		517000	1.4
Landing		517000	1.4
Limit Landing Sinking Speed (ft/sec)			12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)			20
Airframe Weight (as Defined in AN-W-11) (lb)			

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	15	4
Max SL Static Thrust per Engine (lb)	400000	52500
Total Propellant Tankage Volume (ft^3)	129111	–
Total JP Fuel Tankage Volume (ft^3)	–	1027

6. Through 16.	
Maximum Electrical Power (KVA)	23
Pressurized Surface Area (ft^2)	19260
Maximum Mission Duration (days)	7
No. of Crew	2
17. Through 21.	
Maximum Number of Personnel Including Crew	2
Maximum Cargo (lb)	Orbiter
Maximum Cargo Density (lb/ft^3)	—
22. Through 27.	
Capacity Propellant Weight (lb)	2811879
Capacity JP Fuel Weight (lb)	46916
GENERAL/MISSION DESIGN	
Maximum Design Gross Weight (lb)	3402316
Maximum Boost Load Factor 4 g at ________ G. W.	
Maximum Boost Dynamic Pressure (lb/ft^2)	670.2
Main Engine Specific Impulse (lb-sec/lb) (Uprated = 449.5)	389.3
Delta Velocity Available (ft/sec) Ideal	14690
Entry Velocity (ft/sec)	10912
Entry Weight (lb)	590437
Ballistic Coefficient, W/C_DA (lb/ft^2)	–
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 6.9	
Lower Surf = 9.2, Upper Surf = 0.8	
Factors of Safety (Define) See Volume II, Section 2	
Margins (Define)	
Contingency (Define) - Uncertainty Allowance	19487
Weight Growth (Define) - 10% Dry Structure	46987

*℄ Axle to ℄ Trunnion

**Fully Extended to Fully Collapsed

SPACECRAFT SEQUENCE MASS PROPERTIES STATEMENT

Page Of

Configuration
FR-3 POINT DESIGN (15' x 60' P/L BAY)

By

Date

No.	Mission Event	Weight (lb)	Center of Gravity (feet)			Moment of Inertia (slug-ft^2 $\times 10^6$)			Product of Inertia (slug-ft^2 $\times 10^6$)		
			x	y	z	I_{x-x}	I_{y-y}	I_{z-z}	I_{xy}	I_{xz}	I_{yz}
	GROSS LIFT-OFF	4329000	77.8	0	92.5	36.39	356.69	357.39	—	1.039	—
	BOOSTER (l = 210.0)										
	LIFT-OFF	3402316	70.7	0	99.5	8.09	252.14	252.53	—	0.676	—
	MAX. α q	2338760	82.2	0	99.3	6.73	202.28	202.68	—	1.230	—
	BURNOUT	590437	117.3	0	97.4	4.75	98.64	99.14	—	2.940	—
	ENTRY	564247	114.9	0	97.2	4.74	96.02	96.51	—	2.820	—
	FLYBACK (INITIAL)	564247	113.4	0	97.0	9.84	92.90	98.51	—	3.580	—
	LANDING	517331	123.6	0	97.1	9.78	74.66	80.04	—	2.800	—
	BOOSTER PROPELLANTS	2811879	61.3	0	100.0	5.72	107.36	107.36	—	0	—
	MAX. α q PROPELLANTS	1748323	70.3	0	100.0	1.88	73.24	73.24	—	0	—
	FLYBACK PROPELLANTS	46916	8.0	0	92.0	0.002	0.11	0.11	—	0	—
	ORBITER (l = 179.2)										
	LIFT-OFF	926684	73.6	0	99.3	2.78	51.84	52.15	—	0.394	—
	BURNOUT	334444	95.6	0	98.3	2.24	27.80	28.13	—	0.823	—
	ENTRY (PAYLOAD IN)	287873	102.1	0	99.1	2.04	23.55	23.80	—	0.479	—
	FLYBACK (INITIAL)	287873	101.8	0	98.9	3.05	22.66	23.83	—	0.670	—
	LANDING	285005	102.7	0	98.6	3.12	22.29	23.38	—	0.626	—
	PAYLOAD	50000	94.0	0	105.0	0.04	0.49	0.49	—	0	—
	ORBITER PROPELLANTS	592240	61.2	0	99.8	0.52	16.14	16.14	—	NEGL.	—
	FLYBACK PROPELLANT	2868	23.0	0	92.0	NEGL.	NEGL.	NEGL.	—	0	—
			*		**						

NOTES:

* VEHICLE NOSE = STA O (+ AFT)

** GEOMETRIC CENTER LINE = WL 100
GROSS LIFTOFF CG BASED ON BOOSTER REFERENCE SYSTEM

Ref. MIL-M-38310A or SP-6004

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION	BY	DATE
FR-3 POINT DESIGN (15' x 60' P/L BAY)		

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	U
1.0	AERODYNAMIC SURFACES	82798						41084	
2.0	BODY STRUCTURE	191443						69156	
3.0	INDUCED ENVIR PROT	47388						43052	
4.0	LNCH RECOV & DKG	30269						14020	
5.0	MAIN PROPULSION	141261						47539	
6.0	ORIENT CONTROL SEP & ULL	15740						11836	
7.0	PRIME POWER SOURCE	1389						1827	
8.0	POWER CONV & DISTR	4092						2443	
9.0	GUIDANCE & NAVIGATION	336						512	
10.0	INSTRUMENTATION	330						407	
11.0	COMMUNICATION	181						181	
12.0	ENVIRONMENTAL CONTROL	748						1430	
13.0	(RESERVED)								
14.0	PERSONNEL PROVISIONS	297						615	
15.0	CREW STA CONTRL & PAN	308						308	
16.0	RANGE SAFETY & ABORT	275						55	
SUBTOTALS (DRY WEIGHT)		516855						234465	
17.0	PERSONNEL	476						540	
18.0	CARGO							50000	
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	25594						4381	
SUBTOTALS (INERT WEIGHT)		542925						289386	
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	159						4829	
24.0	THRUST DECAY PROPELLANT	437						43	
25.0	FULL THRUST PROPELLANT	2858795						632426	
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
TOTALS (GROSS WEIGHT) (LB)		3402316						926684	
DESIGN ENVELOPE VOLUME (FT^3)		236000						89060	
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)		26610						14900	
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)		670						670	
DESIGN g, MAX		4						4	
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS		2/.1						2/7	

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A - Booster

B

C

D

E

F

SPACECRAFT

M MANNED LAUNCH - Orbiter

U UNMANNED LAUNCH

GROSS WEIGHT 4329000

NOTES & SKETCHES:

Tanks are oversized to account for thrust build-up and preignition losses.

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE ____ OF ____

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 1.0

FOR SYSTEM AERODYNAMIC SURFACE

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
1.1	FIXED SURFACES									16957		
	WING										6426	
	CARRY THROUGH STRUCTURE											3468
	CARRY THROUGH LUGS & BEARINGS											2018
	CARRY THROUGH PIN INSTL.											940
	STABILIZER										9462	
	BASIC STRUCTURE											5853
	SECONDARY STRUCTURE											3609
	INTER-ELEVON FAIRING										1069	
1.2	MOVABLE SURFACES									17169		
	WING										14249	
	OUTER PANEL *											12907
	SPOILERS											141
	FLAPS - TRAILING EDGE											1201
	STABILIZER - RUDDERVATORS **										1565	
	ELEVONS										1355	
1.19	CONTINGENCY									3223		
1.20	GROWTH (10% OF ABOVE)									3735		

	FIRST	SECOND	THIRD
PAGE TOTALS	41084		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	41084		

NOTES:

* INCLUDES PIVOT FTG OF 1328 LBS AND 80 LBS OF TPS

** INCLUDES 0 LBS BALANCE WEIGHTS

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	**DESCRIPTION**	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
2.1	STRUCTURAL FUEL TANK									10470		
	BASIC STRUCTURE										8000	
	SECONDARY STRUCTURE										276	
	INSULATION										2194	
2.2	STRUCTURAL OXIDIZER									7009		
	BASIC STRUCTURE										6900	
	SECONDARY STRUCTURE										109	
2.6	STRUCTURE FWD. OF INTEGRAL TANKS									4430		
	AIR BREATHING ENGINE COMPARTMENT										3991	
	TRANSITION TO LOX TANKS										439	
2.7	STRUCTURE BETWEEN INTEGRAL TANKS									20340		
	CENTER BODY										15000	
	PAYLOAD DOORS										5340	
2.8	STRUCTURE AFT OF INTEGRAL TANKS									3597		
2.9	THRUST STRUCTURE									6088		
2.11	PRESSURIZED COMPARTMENT									2479		
	COCKPIT										1319	
	PERSONNEL TUNNEL										1160	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	54413		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
2.16	STRUCTURE BELOW INTEGRAL TANKS									4846		
	HEAT SHIELD SUPPORT STRUCTURE										3183	
	INTERSTAGE CONNECTIO N PENALTIES										1663	
2.17	STRUCTURE ABOVE INTEGRAL TANKS									1663		
	INTERSTAGE CONNECTION PENALTIES										1663	
2.19	CONTINGENCY									1947		
2.20	GROWTH (10% OF ABOVE)									6287		

NOTES:

PAGE TOTALS	14743		
TOTALS FROM PAGE 1	54413		
SUBTOTALS			
SYSTEM TOTAL	69156		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 3.0

FOR SYSTEM INDUCED ENVIRONMENT PROTECTION

FOR ITEM OR MODULE ORBITER

		UNITS •							TOTALS			
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
3.2	THERMAL PROTECTION (PASSIVE)									37901		
	STABILIZERS										5320	
	ELEVON										835	
	INTER-ELEVON FAIRING										945	
	BODY										30801	
3.8	CONTINGENCY									1237		
3.9	GROWTH									3914		

NOTES:

PAGE TOTALS	4305		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	43052		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 4.0

FOR SYSTEM LAUNCH, RECOVERY & DOCKING

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
4.3	LANDING GEAR									12245		
4.19	CONTINGENCY									500		
4.20	GROWTH									1275		

NOTES:

PAGE TOTALS	14020		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	14020		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
5.1	LIQUID ROCKET ENGINES AND ACCESSORIES									13834		
5.6	AIRBREATHING ENGINES AND ACCESSORIES									16110		
5.8	FUEL CONTAINERS (NON STRUCTURAL)									7310		
	CARGO BAY TANK										2700	
	CARGO BAY TANK SUPPORTS										160	
	CARGO BAY TANK INSULATION										2290	
	ORBIT MANEUVER TANK										990	
	ORBIT MANEUVER TANK SUPPORTS										36	
	ORBIT MANEUVER TANK INSULATION										1134	
5.9	FUEL SYSTEM									1169		
5.11	OXIDIZER CONTAINER (NON STRUCTURAL)									741		
	ORBIT MANEUVER TANK										495	
	ORBIT MANEUVER TANK SUPPORTS										46	
	ORBIT MANEUVER TANK INSULATION										200	
5.12	OXIDIZER SYSTEM									2977		
5.15	PROPELLANT UTILIZATION SYSTEM									125		

NOTES:

PAGE TOTALS	42266		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
5.16	AIRBREATHING FUEL TANKAGE & SYSTEM									265		
5.19	CONTINGENCY									687		
5.20	GROWTH (10% OF ABOVE)									4321		

NOTES:

PAGE TOTALS	5273		
TOTALS FROM PAGE	42266		
SUBTOTALS			
SYSTEM TOTAL	47539		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 6.0

FOR SYSTEM ORIENTATION AND SEPARATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
6.1	THRUST SYSTEM (MAIN)									450		
	GIMBAL SYSTEM										450	
6.2	THRUST SYSTEM (AUXILIARY) (R. C. S.)									1781		
6.3	AERODYNAMIC CONTROL									4233		
	ACTUATORS AND INSTALLATION										1561	
	LINES										1178	
	FLUID										491	
	MECHANISMS										825	
	VALVES										147	
	ELECTRICAL CONTROLS, WIPE, ETC.										31	
6.5	SEPARATION									649		
	BOOSTER - ORBITER										649	
6.7	R. C. S. FUEL CONTAINER									1255		
6.8	R. C. S. OXIDIZER CONTAINER									1210		
6.10	PROPELLANT DISTRIBUTION & CONTROL									679		
	COMPRESSORS										200	
	PLUMBING & SUPPORTS										400	
	VALVES & REGULATORS										79	
6.19	CONTINGENCY									503		
6.20	GROWTH (10% OF ABOVE)									1076		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	11836		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	11836		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 7.0

FOR SYSTEM PRIME POWER SOURCE

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
7.2	POWER SOURCE - FUEL CELL									885		
	FUEL CELLS										294	
	FUEL CONTAINER										257	
	OXIDIZER CONTAINER										259	
	VALVES, HEATERS, PLUMBING										75	
7.3	POWER SOURCE - BATTERIES									120		
7.6	POWER SOURCE - GAS GENERATOR									656		
	HYDRAULIC POWER UNIT										470	
	INSTALLATION										186	
7.20	GROWTH (10% OF ABOVE)									166		

NOTES:

PAGE TOTALS	1827		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	1827		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 8.0

FOR SYSTEM POWER CONVERSION AND DISTRIBUTION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
8.1	POWER CONVERSION - ELECTRICAL A. C.									240		
	AC GENERATORS										150	
	TRANSFORMER - RECTIFIERS										60	
	OIL COOLERS										30	
8.2	POWER CONVERSION - ELECTRICAL D. C.									314		
	ISOLATING RECTIFIERS										8	
	DC CIRCUIT BREAKERS										72	
	CONVERTERS										150	
	GENERATOR WIRING										84	
8.3	POWER CONVERSION - HYDRAULIC/PNEUMATIC									632		
	PUMPS										370	
	RESERVOIRS, HEAT EXCHANGERS										262	
8.4	POWER DISTRIBUTION - ELECTRICAL A. C.									62		
	BUS ENCLOSURE										20	
	AC CIRCUIT BREAKERS										42	
8.5	POWER DISTRIBUTION - ELECTRICAL D. C.									12		
8.6	POWER DISTRIBUTION - HYDRAULIC/PNEUMATIC									961		
	PLUMBING										216	
	FLUID										630	
	VALVES, REGULATORS, ETC.										115	
8.20	GROWTH (10% OF ABOVE)									222		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	2443		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	2443		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 9.0

FOR SYSTEM GUIDANCE AND NAVIGATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
9.1	GUIDANCE SOURCE - REFERENCE									203		
	STAR TRACKER										30	
	RADAR ALTIMETER										20	
	RENDEZVOUS RADAR										120	
	WIRING										15	
	INSTALLATION										18	
9.2	GUIDANCE EVALUATION									262		
	INERTIAL PLATFORM										120	
	COMPUTER										120	
	WIRING										5	
	INSTALLATION										17	
9.20	GROWTH (10% OF ABOVE)									47		

NOTES:

PAGE TOTALS	512		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	512		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 10.0

FOR SYSTEM INSTRUMENTATION

FOR ITEM OR MODULE ORBITER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
10.1	INSTRUMENTATION/ONBOARD CHECKOUT									370		
10.20	GROWTH (10% OF ABOVE)									37		

NOTES:

PAGE TOTALS	407		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	407		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 11.0

FOR SYSTEM COMMUNICATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
11.1	INTERCOMMUNICATIONS									15		
11.2	NEAR-EARTH COMMUNICATION									150		
11.20	GROWTH (10% OF ABOVE)									16		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	181		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	181		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 12.0

FOR SYSTEM ENVIRONMENTAL CONTROL

FOR ITEM OR MODULE ORBITER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
12.2	ECS - PERSONNEL									550		
	ATMOSPHERE CONTROL SYSTEM										154	
	PRESSURIZATION CONTROL SYSTEM										116	
	TEMPERATURE CONTROL SYSTEM										280	
12.3	COOLANT SYSTEM									750		
	RADIATOR PANELS										452	
	RADIATOR CONTROLS										35	
	RADIATOR PLUMBING										20	
	PUMPING UNITS										88	
	HEAT EXCHANGERS										100	
	WATER SUBLIMATORS & HEATERS										36	
	VALVES, XDUCERS, CONTROLS & PLUMBING										19	
12.20	GROWTH (10% OF ABOVE)									130		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	1430		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	1430		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 14.0

FOR SYSTEM PERSONNEL PROVISIONS

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS • CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	TOTALS WEIGHT CODE GENERATION FIRST	SECOND	THIRD
14.1	ACCOMMODATIONS FOR PERSONNEL									250		
	SEATS, SUPTS & RESTRAINTS										250	
14.2	FIXED LIFE SUPPORT EQUIPMENT									189		
	FOOD CONTAINERS										4	
	WATER MANAGEMENT										130	
	WASTE MANAGEMENT										28	
	HYGIENE EQUIPMENT										27	
14.4	FURNISHINGS									100		
14.5	EMERGENCY EQUIPMENT									20		
14.20	GROWTH (10% OF ABOVE)									56		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	615		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	615		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 15.0

FOR SYSTEM CREW STATION CONTROLS & PANELS

FOR ITEM OR MODULE ORBITER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
15.4	CREW STATION CONTROLS									280		
	CONTROLS, PANELS & SUPTS										280	
15.20	GROWTH (10% OF ABOVE)									28		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	308		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	308		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 16.0

FOR SYSTEM RANGE SAFETY & ABORT

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
16.1	RANGE SAFETY & ABORT									50		
16.20	GROWTH (10% OF ABOVE)									5		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	55		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	55		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 17.0

FOR SYSTEM PERSONNEL

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS: CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM: X	Y	Z	QUANTITY	TOTALS: WEIGHT CODE GENERATION: FIRST	SECOND	THIRD
17.1	CREW									450		
	CREW										400	
	PRESSURE SUITS										50	
17.2	PERSONAL GEAR									20		
17.3	LIFE SUPPORT									50		
	FOOD										28	
	WATER										12	
	MEDICAL & FIRST AID										10	
17.4	CREW ACCESSORIES									20		
	MAPS, MANUALS, LOGS										10	
	TOOLS & GEAR										10	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	540		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	540		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 18.0

FOR SYSTEM CARGO

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
18.3	CARGO									50000		

NOTES:

PAGE TOTALS	50000		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	50000		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 21.0

FOR SYSTEM RESIDUAL PROPELLANT & SERVICE ITEMS

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
21.3	FUEL TRAPPED - MAIN ENGINE									918		
	TANK										739	
	ENGINE & LINES										179	
21.4	OXIDIZER TRAPPED - MAIN ENGINE									3037		
	TANK										1871	
	ENGINE & LINES										1166	
21.7	FUEL TRAPPED - ATTITUDE CONTROL									16		
21.8	OXIDIZER TRAPPED - ATTITUDE CONTROL									84		
21.11	FUEL TRAPPED - ELECTRICAL POWER									7		
21.12	OXIDIZER TRAPPED - ELECTRICAL POWER									57		
21.13	SERVICE ITEMS TRAPPED									262		
	ENVIRONMENTAL CONTROL										242	
	LIFE SUPPORT										20	

NOTES:

PAGE TOTALS	4381		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	4381		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 23.0

FOR SYSTEM INFLIGHT LOSSES

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS: CLASS (E, C, A)	UNITS: VOLUME FT^3 EACH	UNITS: POWER WATTS EACH	UNITS: WEIGHT LB EACH	UNITS: LOCATION EACH ORBIT REF. SYSTEM X	UNITS: LOCATION EACH ORBIT REF. SYSTEM Y	UNITS: LOCATION EACH ORBIT REF. SYSTEM Z	QUANTITY	TOTALS: WEIGHT CODE GENERATION FIRST	TOTALS: WEIGHT CODE GENERATION SECOND	TOTALS: WEIGHT CODE GENERATION THIRD
23.3	FUEL - R.C.S.									238		
23.4	OXIDIZER - R.C.S.									1188		
23.5	FUEL - ELECTRICAL SYSTEM									205		
23.6	OXIDIZER - ELECTRICAL SYSTEM									1841		
23.7	SERVICE ITEMS									1357		
	ENVIRONMENTAL CONTROL										25	
	LIFE SUPPORT										1332	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	4829		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	4829		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 24.0

FOR SYSTEM THRUST DECAY PROPELLANT

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
24.1	FUEL									6		
24.2	OXIDIZER									36		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	42		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	42		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 25.0

FOR SYSTEM FULL THRUST PROPELLANT

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	TOTALS WEIGHT CODE GENERATION FIRST	SECOND	THIRD
25.1	FUEL									85185		
	MAIN IMPULSE										78965	
	ORBIT MANEUVER										6220	
25.2	OXIDIZER									544373		
	MAIN IMPULSE										513275	
	ORBIT MANEUVER										31098	
25.4	JET FUEL - AIR BREATHING ENGINES									2868		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	632426		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	632426		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 1.0

FOR SYSTEM AERODYNAMIC SURFACE

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
1.1	FIXED SURFACES											
	WING									13044		
	CARRY THROUGH STRUCTURE										7040	
	CARRY THROUGH LUGS, BEARING										4092	
	CARRY THROUGH PIN INSTL.										1912	
	STABILIZER									20314		
	BASIC STRUCTURE										14566	
	SECONDARY STRUCTURE										5748	
	FIXED ELEVON TYPE SURFACE									0		
1.2	MOVABLE SURFACE											
	WING									28928		
	OUTER PANEL (INCL PIVOT FTG 1328 LBS, TIPS 80 LBS										26202	
	SPOILERS										286	
	FLAP - TRAILING EDGE										2440	
	STABILIZER									3978		
	RUDDERVATORS (INCLUDES BALANCE WEIGHTS 0 LBS)											
	ADJUSTABLE ELEVON TYPE SURFACE									0		
1.19	CONTINGENCY									9007		
1.20	GROWTH (10% OF ABOVE)									7527		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	82798		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	82798		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	TOTALS WEIGHT CODE GENERATION FIRST	SECOND	THIRD
2.1	STRUCTURAL FUEL TANK									59399		
	BASIC STRUCTURE										50126	
	SECONDARY STRUCTURE										1000	
	INSULATION										8273	
2.2	STRUCTURAL OXIDIZER TANK									25307		
	BASIC STRUCTURE										24900	
	SECONDARY STRUCTURE										407	
2.6	STRUCTURE FWD OF INTEGRAL TANKS									15001		
	AIR BREATHING ENGINE COMPARTMENT										12000	
	TRANSITION TO LOX TANK										3001	
2.7	STRUCTURE BETWEEN INTEGRAL TANKS									10100		
2.8	STRUCTURE AFT OF INTEGRAL TANKS									8978		
2.9	THRUST STRUCTURE									27194		
2.11	PRESSURIZED COMPARTMENT									2800		
2.13	BALLAST									9000		

NOTES:

PAGE TOTALS	157779		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS: CLASS (E, C, A)	UNITS: VOLUME FT3 EACH	UNITS: POWER WATTS EACH	UNITS: WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM: X	LOCATION EACH ORBIT REF. SYSTEM: Y	LOCATION EACH ORBIT REF. SYSTEM: Z	QUANTITY	TOTALS: WEIGHT CODE GENERATION: FIRST	TOTALS: WEIGHT CODE GENERATION: SECOND	TOTALS: WEIGHT CODE GENERATION: THIRD
2.16	STRUCTURE BELOW INTEGRAL TANKS									12757		
	HEAT SHIELD SUPPORT STRUCTURE										9000	
	INTERSTAGE CONNECTION PENALTIES										3757	
2.19	CONTINGENCY									3503		
2.20	GROWTH									17404		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	33664		
TOTALS FROM PAGE 1	157779		
SUBTOTALS			
SYSTEM TOTAL	191443		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 3.0

FOR SYSTEM INDUCED ENVIRONMENT PROTECTION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	**DESCRIPTION**	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
3.2	THERMAL PROTECTION (PASSIVE)									41000		
3.8	CONTINGENCY									2080		
3.9	GROWTH									4308		

NOTES:

PAGE TOTALS	47388		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	47388		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 4.0

FOR SYSTEM LAUNCH, RECOVERY & DOCKING

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
4.3	LAUNCH GEAR									2684		
4.1	LANDING GEAR									20800		
4.19	CONTINGENCY									4033		
4.20	GROWTH									2756		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	30269		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	30269		

MSC Form 1522 (Jul 68)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
5.1	LIQUID ROCKET ENGINES & ACCESSORIES									58485		
5.6	AIRBREATHING ENGINES & ACCESSORIES									39600		
5.9	FUEL SYSTEM									6823		
5.12	OXIDIZER SYSTEM									19311		
5.15	PROPELLANT UTILIZATION SYSTEM									150		
5.16	AIRBREATHING FUEL TANKAGE & SYSTEM									3569		
5.19	CONTINGENCY									481		
5.20	GROWTH (10% OF ABOVE)									12842		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	141261		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	141261		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 6.0

FOR SYSTEM ORIENTATION AND SEPARATION

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	WEIGHT CODE GENERATION FIRST	SECOND	THIRD
6.1	THRUST SYSTEM (MAIN)									2000		
	GIMBAL SYSTEM										2000	
6.2	THRUST SYSTEM (AUXILIARY) (R. C. S)									100		
6.3	AERODYNAMIC CONTROL									8308		
	ACTUATORS AND INSTALLATION										3050	
	LINES										1851	
	FLUID										1112	
	MECHANISMS										1886	
	VALVES										339	
	ELECTRICAL CONTROLS, WIRE, ETC.										70	
6.5	SEPARATION									3243		
	BOOSTER - ORBITER										3243	
6.7	R. C. S. FUEL CONTAINER									77		
6.8	R. C. S. OXIDIZER CONTAINER									73		
6.10	PROPELLANT DISTRIBUTION & CONTROL									125		
6.19	CONTINGENCY									383		
6.20	GROWTH (10% OF ABOVE)									1431		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	15740		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	15740		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 7.0

FOR SYSTEM PRIME POWER SOURCE

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	WEIGHT CODE GENERATION FIRST	SECOND	THIRD
7.3	POWER SOURCE - BATTERIES									30		
7.6	POWER SOURCE - GAS GENERATOR									1233		
	HYDRAULIC POWER UNIT										878	
	INSTALLATION										355	
7.20	GROWTH (10% OF ABOVE)									126		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	1389		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	1389		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 8.0

FOR SYSTEM POWER CONVERSION AND DISTRIBUTION

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
		CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
CODE	DESCRIPTION					X	Y	Z		FIRST	SECOND	THIRD
8.1	POWER CONVERSION - ELECTRICAL A. C.									240		
	AC GENERATORS										150	
	TRANSFORMER - RECTIFIERS										60	
	OIL COOLERS										30	
8.2	POWER CONVERSION - ELECTRICAL D. C.									310		
	ISOLATING RECTIFIERS										4	
	DC CIRCUIT BREAKERS										72	
	CONVERTERS										150	
	GENERATOR WIRING										84	
8.3	POWER CONVERSION - HYDRAULIC/PNEUMATIC									1204		
	PUMPS										707	
	RESERVOIRS, HEAT EXCHANGERS										497	
8.4	POWER DISTRIBUTION - ELECTRICAL A. C.									62		
	BUS ENCLOSURE										20	
	AC CIRCUIT BREAKERS										42	
8.5	POWER DISTRIBUTION - ELECTRICAL D. C.									4		
8.6	POWER DISTRIBUTION - HYDRAULIC/PNEUMATIC									1900		
	PLUMBING										442	
	FLUID										1225	
	VALVES, REGULATORS, ETC.										233	
8.20	GROWTH (10% OF ABOVE)									372		

NOTES:

PAGE TOTALS	4092		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	4092		

MSC Form 1522 (Jul 68)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 9.0

FOR SYSTEM GUIDANCE AND NAVIGATION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
9.1	GUIDANCE SOURCE - REFERENCE									20		
	RADAR ALTIMETER										20	
9.2	GUIDANCE - EVALUATION									285		
	INERTIAL PLATFORM										120	
	COMPUTER										120	
	WIRING										20	
	INSTALLATION										25	
9.20	GROWTH (10% OF ABOVE)									31		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	336		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	336		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 10.0

FOR SYSTEM INSTRUMENTATION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT, CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
10.1	INSTRUMENTATION / ONBOARD CHECKOUT									300		
10.20	GROWTH (10% OF ABOVE)									30		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	330		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	330		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 11.0

FOR SYSTEM COMMUNICATION

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
11.1	INTERCOMMUNICATIONS									15		
11.2	NEAR-EARTH COMMUNICATION									150		
11.20	GROWTH (10% OF ABOVE)									16		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	181		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	181		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 12.0

FOR SYSTEM ENVIRONMENTAL CONTROL

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
12.2	ECS - PERSONNEL									550		
	ATMOSPHERIC CONTROL SYSTEM										154	
	PRESSURIZATION CONTROL SYSTEM										116	
	TEMPERATURE CONTROL SYSTEM										280	
12.3	COOLANT SYSTEM									130		
	HEAT EXCHANGERS										11	
	PUMPING UNITS										60	
	BLOWERS										20	
	PLUMBING VALVES										15	
	CONTAINERS										24	
12.20	GROWTH (10% OF ABOVE)									68		

NOTES:

PAGE TOTALS	748		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	748		

MSC Form 1522 (Jul 68)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 14.0

FOR SYSTEM PERSONNEL PROVISIONS

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
14.1	ACCOMMODATIONS FOR PERSONNEL									250		
	SEATS, SUPTS & RESTRAINTS										250	
14.5	EMERGENCY EQUIPMENT									20		
14.20	GROWTH (10% OF ABOVE)									27		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	297		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	297		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 15.0

FOR SYSTEM CREW STATION CONTROLS & PANELS

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
15.4	CREW STATION CONTROLS									280		
	CONTROLS, PANELS & SUPTS										280	
15.20	GROWTH (10% OF ABOVE)									28		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	308		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	308		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 16.0

FOR SYSTEM RANGE SAFETY & ABORT

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
16.1	RANGE SAFETY & ABORT									250		
16.20	GROWTH (10% OF ABOVE)									25		

NOTES:

PAGE TOTALS	275		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	275		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 17.0

FOR SYSTEM PERSONNEL

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
17.1	CREW									450		
	CREW										400	
	PRESSURE SUITS										50	
17.3	LIFE SUPPORT									16		
17.4	CREW ACCESSORIES									10		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	476		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	476		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 21.0

FOR SYSTEM RESIDUAL PROPELLANT & SERVICE ITEMS

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
21.3	FUEL TRAPPED - MAIN ENGINE									5951		
	TANK										4133	
	ENGINE & LINES										1818	
21.4	OXIDIZER TRAPPED - MAIN ENGINE									19513		
	TANK										7696	
	ENGINE & LINES										11817	
21.13	SERVICE ITEMS TRAPPED									130		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	25594		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	25594		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 23.0

FOR SYSTEM INFLIGHT LOSSES

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
23.5	FUEL - ELECTRICAL SYSTEM									16		
23.6	OXIDIZER - ELECTRICAL SYSTEM									143		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	159		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	159		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 24.0

FOR SYSTEM THRUST DECAY PROPELLANT

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	WEIGHT CODE GENERATION FIRST	SECOND	THIRD
24.1	FUEL									58		
24.2	OXIDIZER									379		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	437		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	437		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-3 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 25.0

FOR SYSTEM FULL THRUST PROPELLANT

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
25.1	FUEL									374917		
25.2	OXIDIZER									2436962		
25.4	JET FUEL - AIR BREATHING ENGINES									46916		

NOTES:

PAGE TOTALS	2858795		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	2858795		

MSC Form 1522 (Jul 69)

11.2 FR-4 CONFIGURATION

The Spacecraft Design Data Statement appears on the first two pages of the following compilation of mass property data for the FR-4 configuration. The Spacecraft Sequence Mass Property Statement and the Spacecraft Summary Weight Statement are on the next two pages. These are followed by the Spacecraft Detail Weight Statements for the FR-4 configuration.

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-4 ORBITER (15' x 60' P/L BAY)

BY

DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces			Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)			–		–
Surface Area of Above Volume (ft^2)			–		–
Gross Area (ft^2)			1514.0		1583.0
Span (ft)			151.3		69.5
Mean Aerodynamic Chord Length (ft)			13.0		–
Theoretical Root Chord Length (in.)			172.2		359.0
Theoretical Root Chord Max. Thickness (in.)			36.2		43.1
Theoretical Tip Chord Length (in.)			137.3		215.5
Theoretical Tip Chord Max. Thickness (in.)			24.7		21.5
2. Body Structure		Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)			10638	107470	
Pressurized Volume (ft^3)		350			29219
Surface Area, Wetted, (ft^2)				16890	
Maximum Length (ft)				190.8	
Maximum Depth (ft)				28.1	
Maximum Width (ft)				32.9	
Total Wetted Outer Mold Line Volume (ft^3)				107470	
Total Wetted Outer Mold Line Surface Area (ft^2)				16890	
3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)			660	3280	358
Surface Area Delta				16890	
Within 1.0 & 2.0 (ft^2)			3180		4294
Windward Unit Wt. (lb/ft^2)			2.4	2.78	
Leeward Unit Wt. (lb/ft^2)			1.88	2.13	
4. Launch Recovery and Docking			Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)			210000	90000	
Length, Oleo Extended (in.)*			180	180	
Oleo Travel (in.)**			30	36	
Structural				Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)				322000	1.4
Landing				322000	1.4
Limit Landing Sinking Speed (ft/sec)					12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)					20
Airframe Weight (as Defined in AN-W-11) (lb)					

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	3	2
Max SL Static Thrust per Engine (lb)	400000	40600
Total Propellant Tankage Volume (ft^3)	29219	–
Total JP Fuel Tankage Volume (ft^3)	–	73
6. Through 16.		
Maximum Electrical Power (KVA)		23
Pressurized Surface Area (ft^2)		6537
Maximum Mission Duration (days)		7
No. of Crew		2
17. Through 21.		
Maximum Number of Personnel Including Crew		2
Maximum Cargo (lb)		50000
Maximum Cargo Density (lb/ft^3)		4.7
22. Through 27.		
Capacity Propellant Weight (lb)		825351
Capacity JP Fuel Weight (lb)		3225
GENERAL/MISSION DESIGN		
Maximum Design Gross Weight (lb)		1160996
Maximum Boost Load Factor 4 g at 1160996 G. W.		
Maximum Boost Dynamic Pressure (lb/ft^2)		657.8
Main Engine Specific Impulse (lb-sec/lb) (Vacuum)		459.0
Delta Velocity Available (ft/sec) Ideal		29622
Entry Velocity (ft/sec) Altitude = 400000 Ft $\gamma = -1.0^{\circ}$		25970
Entry Weight (lb)		327176
Ballistic Coefficient, W/C_DA (lb/ft^2)		–
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 58.6		
Lower Surf = 11.3, Upper Surf = 0.35, Body L. E. = 46.3		
Factors of Safety (Define) See Volume II, Section 2		–
Margins (Define)		–
Contingency (Define) - Uncertainty Allowance		10109
Weight Growth (Define) - 10% Dry Structure		24694

*℄ Axle to ℄ Trunnion

**Fully Extended to Fully Collapsed

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-4 BOOSTER (15' x 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces	Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)	–		–
Surface Area of Above Volume (ft^2)	–		–
Gross Area (ft^2) (Exposed)	1607.0		1728.0
Span (ft)	158.3		73.0
Mean Aerodynamic Chord Length (ft)	13.6		–
Theoretical Root Chord Length (in.)	205.0		374.0
Theoretical Root Chord Max. Thickness (in.)	41.0		44.8
Theoretical Tip Chord Length (in.)	143.5		225.0
Theoretical Tip Chord Max. Thickness (in.)	25.8		22.5

2. Body Structure	Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)			122370	
Pressurized Volume (ft^3)	350			69306
Surface Area, Wetted, (ft^2)			18420	
Maximum Length (ft)			199.3	
Maximum Depth (ft)			29.4	
Maximum Width (ft)			34.4	
Total Wetted Outer Mold Line Volume (ft^3)			122370	
Total Wetted Outer Mold Line Surface Area (ft^2)			18420	

3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within 1.0 & 2.0 (ft^3)				2300	590.0
Surface Area Delta				18420	
Within 1.0 & 2.0 (ft^2)					7075.0
Windward Unit Wt. (lb/ft^2)				2.52	
Leeward Unit Wt. (lb/ft^2)				1.16	

4. Launch Recovery and Docking	Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)	210000	90000	
Length, Oleo Extended (in.)*	180	180	
Oleo Travel (in.)**	30	36	
Structural		Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)		325000	1.4
Landing		325000	1.4
Limit Landing Sinking Speed (ft/sec)			12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)			20
Airframe Weight (as Defined in AN-W-11) (lb)			

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	9	3
Max SL Static Thrust per Engine (lb)	400000	40600
Total Propellant Tankage Volume (ft^3)	69306	–
Total JP Fuel Tankage Volume (ft^3)	–	695

6. Through 16.	
Maximum Electrical Power (KVA)	23
Pressurized Surface Area (ft^2)	9825
Maximum Mission Duration (days)	1.5 hr
No. of Crew	2
17. Through 21.	
Maximum Number of Personnel Including Crew	2
Maximum Cargo (lb)	Orbiter
Maximum Cargo Density (lb/ft^3)	–
22. Through 27.	
Capacity Propellant Weight (lb)	1508915
Capacity JP Fuel Weight (lb)	30717
GENERAL/MISSION DESIGN	
Maximum Design Gross Weight (lb) (Each)	1879048
Maximum Boost Load Factor 4 g at ______ G. W.	
Maximum Boost Dynamic Pressure (lb/ft^2)	657.8
Main Engine Specific Impulse (lb-sec/lb). (Uprated = 449.5)	389.3
Delta Velocity Available (ft/sec) Ideal	13268
Entry Velocity (ft/sec)	9411
Entry Weight (lb)	355767
Ballistic Coefficient, W/C_DA (lb/ft^2)	–
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 8.5	
Factors of Safety (Define) See Volume II, Section 2	–
Margins (Define)	
Contingency (Define) - Uncertainty Allowance	11703
Weight Growth (Define) - 10% Dry Structure	29495

*℄ Axle to ℄ Trunnion

**Fully Extended to Fully Collapsed

SPACECRAFT SEQUENCE MASS PROPERTIES STATEMENT

Page Of

Configuration
FR-4 POINT DESIGN (15' x 60' P/L BAY)

By

Date

No.	Mission Event	Weight (lb)	Center of Gravity (feet) x	y	z	Moment of Inertia (slug-ft^2 × 10^6) I_{x-x}	I_{y-y}	I_{z-z}	Product of Inertia (slug-ft^2 × 10^6) I_{xy}	I_{xz}	I_{yz}
	GROSS LIFT-OFF	4919092	84.0	0	99.3	130.54	408.95	289.94	—	1.049	—
	BOOSTER (l = 199.3)										
	LIFT-OFF	1879048	85.9	0	99.5	3.84	108.85	108.97	—	0.408	—
	MAX. α q	1249786	98.4	0	99.3	3.36	80.52	80.65	—	0.769	—
	BURNOUT	370133	122.1	0	97.5	2.39	46.96	47.14	—	1.460	—
	ENTRY	355636	121.3	0	97.4	2.38	45.97	46.15	—	1.430	—
	FLYBACK (INITIAL)	355636	119.8	0	97.2	3.66	44.38	45.73	—	1.900	—
	LANDING	324926	128.4	0	97.3	3.62	36.37	37.59	—	1.400	—
	BOOSTER PROPELLANTS	1508915	81.0	0	100.0	2.38	43.69	43.69	—	0	—
	MAX. α q PROPELLANTS	879653	88.4	0	100.0	0.91	24.33	24.33	—	0	—
	FLYBACK PROPELLANT	30711	40.0	0	92.0	0.001	0.03	0.03	—	0	—
	ORBITER										
	LIFT-OFF (l = 190.8)	1160996	78.0	0	99.2	3.32	69.98	70.27	—	0.119	—
	BURNOUT	377763	104.3	0	98.1	2.61	32.47	32.79	—	0.804	—
	ENTRY (PAYLOAD IN)	325393	108.7	0	98.8	2.39	29.72	29.95	—	0.559	—
	FLYBACK (INITIAL)	325393	107.4	0	98.7	3.65	28.33	29.74	—	0.889	—
	LANDING	322168	108.3	0	98.4	3.73	27.78	29.10	—	0.817	—
	PAYLOAD	50000	97.0	0	105.0	0.04	0.49	0.49	—	0	—
	ORBITER PROPELLANTS	783233	65.4	0	99.8	0.69	25.50	25.50	—	- 0.159	—
	FLYBACK PROPELLANT	3225	23.0	0	92.0	0	0	0	—	0	—
			*		**						

NOTES:
- * VEHICLE NOSE = STA O (+ AFT)
- ** GEOMETRIC CENTERLINE = WL 100 (+ UP)

Ref. MIL-M-38310A or SP-6004

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY:

DATE:

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	U
1.0	AERODYNAMIC SURFACES	52988	52988					46608	
2.0	BODY STRUCTURE	114428	114428					82709	
3.0	INDUCED ENVIR PROT	36092	36092					55361	
4.0	LNCH RECOV & DKG	15835	15835					15830	
5.0	MAIN PROPULSION	88464	88464					49717	
6.0	ORIENT CONTROL SEP & ULL	10603	10603					13304	
7.0	PRIME POWER SOURCE	908	908					1921	
8.0	POWER CONV & DISTR	2656	2656					2670	
9.0	GUIDANCE & NAVIGATION	336	336					512	
10.0	INSTRUMENTATION	330	330					407	
11.0	COMMUNICATION	181	181					181	
12.0	ENVIRONMENTAL CONTROL	748	748					1430	
13.0	(RESERVED)								
14.0	PERSONNEL PROVISIONS	297	297					615	
15.0	CREW STA CONTRL & PAN	308	308					308	
16.0	RANGE SAFETY & ABORT	275	275					55	
SUBTOTALS (DRY WEIGHT)		324449	324449					271628	
17.0	PERSONNEL	476	476					540	
18.0	CARGO							50000	
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	14109	14109					5195	
SUBTOTALS (INERT WEIGHT)		339034	339034					327363	
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	159	159					5014	
24.0	THRUST DECAY PROPELLANT	229	229					43	
25.0	FULL THRUST PROPELLANT	1539626	1539626					828576	
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
TOTALS (GROSS WEIGHT) (LB)		1879048	1879048					1160996	
DESIGN ENVELOPE VOLUME (FT^3)		122370	122370					107470	
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA - BODY (FT^2)		18420	18420					16890	
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)		658	658					658	
DESIGN g, MAX.		4	4					4	
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS		2/.1	2/.1					2/7	

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A - Booster

B - Booster

C

D

E

F

SPACECRAFT

M MANNED LAUNCH - Orbiter

U UNMANNED LAUNCH

GROSS WEIGHT 4919092

NOTES & SKETCHES:

Tanks are over-sized to account for thrust build-up and preignition losses

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 1.0

FOR SYSTEM AERODYNAMIC SURFACE

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	WEIGHT CODE GENERATION FIRST	SECOND	THIRD
1.1	FIXED SURFACES											
	WING									7290		
	CARRY THROUGH STRUCTURE										3934	
	CARRY THROUGH LUGS, BEARING										2287	
	CARRY THROUGH PIN INSTL.										1069	
	STABILIZER									10734		
	BASIC STRUCTURE										6640	
	SECONDARY STRUCTURE										4094	
	INTER-ELEVON FAIRING									1213		
1.2	MOVABLE SURFACE											
	WING									16164		
	OUTER PANEL (INCL PIVOT FTG 1328 LBS, TIPS 80 LBS)										14641	
	SPOILERS										160	
	FLAP - TRAILING EDGE										1363	
	STABILIZER									1776		
	RUDDERVATORS (INCL BALANCE WEIGHTS 0 LBS)										1776	
	ELEVONS									1538		
1.19	CONTINGENCY									3656		
1.20	GROWTH (10% OF ABOVE)									4237		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	46608		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	46608		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE ORBITER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
2.1	STRUCTURAL FUEL TANK									13115		
	BASIC STRUCTURE										10200	
	SECONDARY STRUCTURE										339	
	INSULATION										2576	
2.2	STRUCTURAL OXIDIZER									9269		
	BASIC STRUCTURE										9010	
	SECONDARY STRUCTURE										259	
2.6	STRUCTURE FWD OF INTEGRAL TANKS									5665		
	AIR BREATHING ENGINE COMPARTMENT										5104	
	TRANSITION TO LOX TANK										561	
2.7	STRUCTURE BETWEEN INTEGRAL TANKS									22292		
	CENTER BODY										16952	
	PAYLOAD DOORS										5340	
2.8	STRUCTURE AFT OF INTEGRAL TANKS									4600		
2.9	THRUST STRUCTURE									6088		
2.11	PRESSURIZED COMPARTMENT									2479		
	COCKPIT										1319	
	PERSONNEL TUNNEL										1160	

NOTES:

PAGE TOTALS	63508		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
2.16	STRUCTURE BELOW INTEGRAL BANKS									7061		
	HEAT SHIELD SUPPORT STRUCTURE										4934	
	INTERSTAGE CONNECTION PENALTIES										2127	
2.17	STRUCTURE ABOVE INTEGRAL TANK									2127		
	INTERSTAGE CONNECTION PENALTIES										2127	
2.19	CONTINGENCY									2494		
2.20	GROWTH									7519		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	19201		
TOTALS FROM PAGE 1	63509		
SUBTOTALS			
SYSTEM TOTAL	82709		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 3.0

FOR SYSTEM INDUCED ENVIRONMENT PROTECTION

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	TOTALS WEIGHT CODE GENERATION FIRST	SECOND	THIRD
3.2	THERMAL PROTECTION (PASSIVE)									48438		
	STABILIZERS										6816	
	ELEVON										1071	
	INTER-ELEVON FAIRING										961	
	BODY										39590	
3.8	CONTINGENCY									1890		
3.9	GROWTH									5033		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	55361		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	55361		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 4.0

FOR SYSTEM LAUNCH RECOVERY & DOCKING

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
4.3	LANDING GEAR									13890		
4.19	CONTINGENCY									500		
4.20	GROWTH									1440		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	15830		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	15830		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 2

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
5.1	LIQUID ROCKET ENGINES AND ACCESSORIES									13835		
5.6	AIRBREATHING ENGINES AND ACCESSORIES									15292		
5.8	FUEL CONTAINERS (NON STRUCTURAL)									9651		
	CARGO BAY TANK										4815	
	CARGO BAY TANK SUPPORTS										250	
	CARGO BAY TANK INSULATION										2290	
	ORBIT MANEUVER TANK										1080	
	ORBIT MANEUVER TANK SUPPORTS										82	
	ORBIT MANEUVER TANK INSULATION										1134	
5.9	FUEL SYSTEM									1187		
5.11	OXIDIZER CONTAINER (NON STRUCTURAL)									807		
	ORBIT MANEUVER TANK										540	
	ORBIT MANEUVER TANK SUPPORTS										67	
	ORBIT MANEUVER TANK INSULATION										200	
5.12	OXIDIZER SYSTEM									3003		
5.15	PROPELLANT UTILIZATION SYSTEM									125		

NOTES:

PAGE TOTALS	43900		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS							TOTALS			
		CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
5.16	AIRBREATHING FUEL TANKAGE & SYSTEM									292		
5.19	CONTINGENCY									1005		
5.20	GROWTH (10% OF ABOVE)									4520		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	5817		
TOTALS FROM PAGE 1	43900		
SUBTOTALS			
SYSTEM TOTAL	49717		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 6.0

FOR SYSTEM ORIENTATION AND SEPARATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
6.1	THRUST SYSTEM (MAIN)									600		
	GIMBAL SYSTEM										600	
6.2	THRUST SYSTEM (AUXILIARY) (R.C.S.)									1978		
6.3	AERODYNAMIC CONTROL									4766		
	ACTUATORS AND INSTALLATION										1759	
	LINES										1330	
	FLUID										549	
	MECHANISMS										932	
	VALVES										161	
	ELECTRICAL CONTROLS, WIRE, ETC.										35	
6.5	SEPARATION									813		
	BOOSTER - ORBITER										813	
6.7	R.C.S. FUEL CONTAINER									1335		
6.8	R.C.S. OXIDIZER CONTAINER									1288		
6.10	PROPELLANT DISTRIBUTION & CONTROL									751		
	COMPRESSORS										200	
	PLUMBING & SUPPORTS										431	
	VALVES & REGULATORS										120	
6.19	CONTINGENCY									564		
6.20	GROWTH (10% OF ABOVE)									1209		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	13304		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	13304		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 7.0

FOR SYSTEM PRIME POWER SOURCE

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
7.2	POWER SOURCE - FUEL CELL									940		
	FUEL CELLS										314	
	FUEL CONTAINER										272	
	OXIDIZER CONTAINER										274	
	VALVES, HEATERS, PLUMBING										80	
7.3	POWER SOURCE - BATTERIES									120		
7.6	POWER SOURCE - GAS GENERATOR									686		
	HYDRAULIC POWER UNIT										490	
	INSTALLATION										196	
7.20	GROWTH (10% OF ABOVE)									175		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	1921		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	1921		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 8.0

FOR SYSTEM POWER CONVERSION AND DISTRIBUTION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
8.1	POWER CONVERSION - ELECTRICAL A. C.									240		
	AC GENERATORS										150	
	TRANSFORMER - RECTIFIERS										60	
	OIL COOLERS										30	
8.2	POWER CONVERSION - ELECTRICAL D. C.									314		
	ISOLATING RECTIFIERS										8	
	DC CIRCUIT BREAKERS										72	
	CONVERTERS										150	
	GENERATOR WIRING										84	
8.3	POWER CONVERSION - HYDRAULIC/PNEUMATIC									698		
	PUMPS										410	
	RESERVOIRS, HEAT EXCHANGERS										288	
8.4	POWER DISTRIBUTION - ELECTRICAL A. C.									62		
	BUS ENCLOSURE										20	
	AC CIRCUIT BREAKERS										42	
8.5	POWER DISTRIBUTION - ELECTRICAL D. C.									12		
8.6	POWER DISTRIBUTION - HYDRAULIC/PNEUMATIC									1101		
	PLUMBING										256	
	FLUID										710	
	VALVES, REGULATORS, ETC.										135	
8.20	GROWTH (10% OF ABOVE)									243		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	2670		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	2670		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 9.0

FOR SYSTEM GUIDANCE AND NAVIGATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
9.1	GUIDANCE SOURCE - REFERENCE									203		
	STAR TRACKER										30	
	RADAR ALTIMETER										20	
	RENDEZVOUS RADAR										120	
	WIRING										15	
	INSTALLATION										18	
9.2	GUIDANCE EVALUATION									262		
	INERTIAL PLATFORM										120	
	COMPUTER										120	
	WIRING										5	
	INSTALLATION										17	
9.20	GROWTH (10% OF ABOVE)									47		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	512		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	512		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 10.0

FOR SYSTEM INSTRUMENTATION

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	WEIGHT CODE GENERATION FIRST	SECOND	THIRD
10.1	INSTRUMENTATION/ONBOARD CHECKOUT									370		
10.20	GROWTH (10% OF ABOVE)									37		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	407		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	407		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 11.0

FOR SYSTEM COMMUNICATION

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
11.1	INTERCOMMUNICATIONS									15		
11.2	NEAR-EARTH COMMUNICATION									150		
11.20	GROWTH (10% OF ABOVE)									16		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	181		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	181		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 12.0

FOR SYSTEM ENVIRONMENTAL CONTROL

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
12.2	ECS - PERSONNEL									550		
	ATMOSPHERE CONTROL SYSTEM										154	
	PRESSURIZATION CONTROL SYSTEM										116	
	TEMPERATURE CONTROL SYSTEM										280	
12.3	COOLANT SYSTEM									750		
	RADIATOR PANELS										452	
	RADIATOR CONTROLS										35	
	RADIATOR PLUMBING										20	
	PUMPING UNITS										88	
	HEAT EXCHANGERS										100	
	WATER SUBLIMATORS & HEATERS										36	
	VALVES, XDUCERS, CONTROLS & PLUMBING										19	
12.20	GROWTH (10% OF ABOVE)									130		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	1430		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	1430		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 14.0

FOR SYSTEM PERSONNEL PROVISIONS

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
14.1	ACCOMMODATIONS FOR PERSONNEL									250		
	SEATS, SUPTS & RESTRAINTS										250	
14.2	FIXED LIFE SUPPORT EQUIPMENT									189		
	FOOD CONTAINERS										4	
	WATER MANAGEMENT										130	
	WASTE MANAGEMENT										28	
	HYGIENE EQUIPMENT										27	
14.4	FURNISHINGS									100		
14.5	EMERGENCY EQUIPMENT									20		
14.20	GROWTH (10% OF ABOVE)									56		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	615		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	615		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 15.0

FOR SYSTEM CREW STATION CONTROLS & PANELS

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
CODE	DESCRIPTION					X	Y	Z		FIRST	SECOND	THIRD
15.4	CREW STATION CONTROLS									280		
	CONTROLS, PANELS & SUPTS										280	
15.20	GROWTH (10% OF ABOVE)									28		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	308		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	308		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 16.0

FOR SYSTEM RANGE SAFETY AND ABORT

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
		CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
CODE	DESCRIPTION					X	Y	Z		FIRST	SECOND	THIRD
16.1	RANGE SAFETY & ABORT									50		
16.20	GROWTH (10% OF ABOVE)									5		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	55		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	55		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 17.0

FOR SYSTEM PERSONNEL

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
17.1	CREW									450		
	CREW										400	
	PRESSURE SUITS										50	
17.2	PERSONAL GEAR									20		
17.3	LIFE SUPPORT									50		
	FOOD										28	
	WATER										12	
	MEDICAL & FIRST AID										10	
17.4	CREW ACCESSORIES									20		
	MAPS, MANUALS, LOGS										10	
	TOOLS & GEAR										10	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	540		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	540		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 18.0

FOR SYSTEM CARGO

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
18.3	CARGO									50000		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	50000		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	50000		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE ______ OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 21.0

FOR SYSTEM RESIDUAL PROPELLANT & SERVICE ITEMS

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
21.3	FUEL TRAPPED - MAIN ENGINE									1128		
	TANK										949	
	ENGINE & LINES										179	
21.4	OXIDIZER TRAPPED - MAIN ENGINE									3641		
	TANK										2475	
	ENGINE & LINES										1166	
21.7	FUEL TRAPPED - ATTITUDE CONTROL									16		
21.8	OXIDIZER TRAPPED - ATTITUDE CONTROL									84		
21.11	FUEL TRAPPED - ELECTRICAL POWER									7		
21.12	OXIDIZER TRAPPED - ELECTRICAL POWER									57		
21.13	SERVICE ITEMS TRAPPED									262		
	ENVIRONMENTAL CONTROL										242	
	LIFE SUPPORT										20	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	5195		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	5195		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 23.0

FOR SYSTEM INFLIGHT LOSSES

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
23.3	FUEL - R.C.S.									269		
23.4	OXIDIZER - R.C.S.									1342		
23.5	FUEL - ELECTRICAL SYSTEM									205		
23.6	OXIDIZER - ELECTRICAL SYSTEM									1841		
23.7	SERVICE ITEMS									1357		
	ENVIRONMENTAL CONTROL										25	
	LIFE SUPPORT										1332	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	5014		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	5014		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 24.0

FOR SYSTEM THRUST DECAY PROPELLANT

FOR ITEM OR MODULE ORBITER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
24.1	FUEL									7		
24.2	OXIDIZER									36		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	43		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	43		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 25.0

FOR SYSTEM FULL THRUST PROPELLANT

FOR ITEM OR MODULE ORBITER

CODE	DESCRIPTION	UNITS CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM X	Y	Z	QUANTITY	TOTALS WEIGHT CODE GENERATION FIRST	SECOND	THIRD
25.1	FUEL									111451		
	MAIN IMPULSE										104431	
	ORBIT MANEUVER										7020	
25.2	OXIDIZER									713900		
	MAIN IMPULSE										678802	
	ORBIT MANEUVER										35098	
25.4	JET FUEL - AIR BREATHING ENGINES									3225		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	828576		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	828576		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 1.0

FOR SYSTEM AERODYNAMIC SURFACE

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
1.1	FIXED SURFACES											
	WING									7290		
	CARRY THROUGH STRUCTURE										3934	
	CARRY THROUGH LUGS, BEARING										2287	
	CARRY THROUGH PIN INSTL.										1069	
	STABILIZER									14818		
	BASIC STRUCTURE										10625	
	SECONDARY STRUCTURE										4193	
	FIXED ELEVON TYPE SURFACE									1498		
1.2	MOVABLE SURFACE											
	WING									16164		
	OUTER PANEL (INCL PIVOT FTG 1328 LBS, TIPS 80 LBS)										14641	
	SPOILERS										160	
	FLAP - TRAILING EDGE										1363	
	STABILIZER									2902		
	RUDDERVATORS (INCL BALANCE WEIGHTS 0 LBS)										2902	
	ADJUSTABLE ELEVON TYPE SURFACE									1774		
1.19	CONTINGENCY									3725		
1.20	GROWTH (10% OF ABOVE)									4817		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	52988		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	52988		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 2

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS •							TOTALS			
		CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
2.3	STRUCTURAL PROPELLANT TANKS									46237		
	BASIC STRUCTURE										39199	
	SECONDARY STRUCTURE										1500	
	INSULATION - FUEL										5538	
2.6	STRUCTURE FWD OF INTEGRAL TANKS									7400		
	AIR BREATHING ENGINE COMPARTMENT										6700	
	TRANSITION TO LOX TANK										700	
2.8	STRUCTURE AFT OF INTEGRAL TANKS									5971		
2.9	THRUST STRUCTURE									17886		
2.11	PRESSURIZED COMPARTMENT									1319		
	COCKPIT										1319	

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	78813		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL			

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 2 OF 2

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 2.0

FOR SYSTEM BODY STRUCTURE

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
2.13	BALLAST									11000		
2.16	STRUCTURE BELOW INTEGRAL TANKS									6328		
	HEAT SHIELD SUPPORT STRUCTURE										4201	
	INTERSTAGE CONNECTION PENALTIES										2127	
2.17	STRUCTURE ABOVE INTEGRAL TANK									2127		
	INTERSTAGE CONNECTION PENALTIES										2127	
2.19	CONTINGENCY									5757		
2.20	GROWTH									10403		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	38621		
TOTALS FROM PAGE 1	78813		
SUBTOTALS			
SYSTEM TOTAL	114428		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 3.0

FOR SYSTEM INDUCED ENVIRONMENTAL PROTECTION

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS: CLASS (E, C, A)	UNITS: VOLUME FT³ EACH	UNITS: POWER WATTS EACH	UNITS: WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM: X	LOCATION EACH ORBIT REF. SYSTEM: Y	LOCATION EACH ORBIT REF. SYSTEM: Z	QUANTITY	TOTALS: WEIGHT CODE GENERATION: FIRST	TOTALS: WEIGHT CODE GENERATION: SECOND	TOTALS: WEIGHT CODE GENERATION: THIRD
3.2	THERMAL PROTECTION (PASSIVE)									31711		
	STABILIZERS										0	
	ADJUSTABLE ELEVON TYPE SURFACE										1071	
	FIXED ELEVON TYPE SURFACE										961	
	BODY										29679	
3.8	CONTINGENCY									1100		
3.9	GROWTH									3281		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	36092		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	36092		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 4.0

FOR SYSTEM LAUNCH, RECOVERY & DOCKING

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
4.3	LANDING GEAR									13896		
4.19	CONTINGENCY									500		
4.20	GROWTH									1440		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	15835		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	15835		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 5.0

FOR SYSTEM MAIN PROPULSION

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
5.1	LIQUID ROCKET ENGINES & ACCESSORIES									39417		
5.6	AIRBREATHING ENGINES & ACCESSORIES									22938		
5.9	FUEL SYSTEM									3933		
5.12	OXIDIZER SYSTEM									11269		
5.15	PROPELLANT UTILIZATION SYSTEM									150		
5.16	AIRBREATHING FUEL TANKAGE & SYSTEM									2353		
5.19	CONTINGENCY									362		
5.20	GROWTH (10% OF ABOVE)									8042		

NOTES:

PAGE TOTALS	88464		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	88464		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 6.0

FOR SYSTEM ORIENTATION AND SEPARATION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
6.1	THRUST SYSTEM (MAIN)									900		
	GIMBAL SYSTEM										900	
6.2	THRUST SYSTEM (AUXILIARY) (R.C.S.)									100		
6.3	AERODYNAMIC CONTROL									5220		
	ACTUATORS AND INSTALLATION										1923	
	LINES										1161	
	FLUID										701	
	MECHANISMS										1183	
	VALVES										208	
	ELECTRICAL CONTROLS, WIRE, ETC.										44	
6.5	SEPARATION									2885		
	BOOSTER - ORBITER										2885	
6.7	R.C.S. FUEL CONTAINER									77		
6.8	R.C.S. OXIDIZER CONTAINER									73		
6.10	PROPELLANT DISTRIBUTION & CONTROL									125		
6.19	CONTINGENCY									259		
6.20	GROWTH (10% OF ABOVE)									964		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	10603		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	10603		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 7.0

FOR SYSTEM PRIME POWER SOURCE

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
7.3	POWER SOURCE - BATTERIES									30		
7.6	POWER SOURCE - GAS GENERATOR									795		
	HYDRAULIC POWER UNIT										566	
	INSTALLATION										229	
7.20	GROWTH (10% OF ABOVE)									83		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	908		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	908		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 8.0

FOR SYSTEM POWER CONVERSION AND DISTRIBUTION

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
8.1	POWER CONVERSION - ELECTRICAL A. C.									240		
	AC GENERATORS										150	
	TRANSFORMER - RECTIFIERS										60	
	OIL COOLERS										30	
8.2	POWER CONVERSION - ELECTRICAL D. C.									310		
	ISOLATING RECTIFIERS										4	
	DC CIRCUIT BREAKERS										72	
	CONVERTERS										150	
	GENERATOR WIRING										84	
8.3	POWER CONVERSION - HYDRAULIC/PNEUMATIC									698		
	PUMPS										410	
	RESERVOIRS, HEAT EXCHANGERS										288	
8.4	POWER DISTRIBUTION - ELECTRICAL A. C.									62		
	BUS ENCLOSURE										20	
	AC CIRCUIT BREAKERS										42	
8.5	POWER DISTRIBUTION - ELECTRICAL D. C.									4		
8.6	POWER DISTRIBUTION - HYDRAULIC/PNEUMATIC									1101		
	PLUMBING										256	
	FLUID										710	
	VALVES, REGULATORS, ETC.										135	
8.20	GROWTH (10% OF ABOVE)									241		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	2656		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	2656		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 9.0

FOR SYSTEM GUIDANCE AND NAVIGATION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
9.1	GUIDANCE SOURCE - REFERENCE									20		
	RADAR ALTIMETER										20	
9.2	GUIDANCE EVALUATION									285		
	INERTIAL PLATFORM										120	
	COMPUTER										120	
	WIRING										20	
	INSTALLATION										25	
9.20	GROWTH (10% OF ABOVE)									31		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	336		
TOTALS FROM PAGE ___			
SUBTOTALS			
SYSTEM TOTAL	336		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 10.0

FOR SYSTEM INSTRUMENTATION

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
10.1	INSTRUMENTATION / ONBOARD CHECKOUT									300		
10.20	GROWTH (10% OF ABOVE)									30		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	330		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	330		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 11.0

FOR SYSTEM COMMUNICATION

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF, SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
11.1	INTERCOMMUNICATIONS									15		
11.2	NEAR-EARTH COMMUNICATION									150		
11.20	GROWTH (10% OF ABOVE)									16		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	181		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	181		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 12.0

FOR SYSTEM ENVIRONMENTAL CONTROL

FOR ITEM OR MODULE BOOSTER

CODE	DESCRIPTION	UNITS							TOTALS			
		CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	LOCATION EACH ORBIT REF. SYSTEM			QUANTITY	WEIGHT CODE GENERATION		
						X	Y	Z		FIRST	SECOND	THIRD
12.2	ECS - PERSONNEL									550		
	ATMOSPHERIC CONTROL SYSTEM										154	
	PRESSURIZATION CONTROL SYSTEM										116	
	TEMPERATURE CONTROL SYSTEM										280	
12.3	COOLANT SYSTEM									130		
	HEAT EXCHANGERS										11	
	PUMPING UNITS										60	
	BLOWERS										20	
	PLUMBING, VALVES										15	
	CONTAINERS										24	
12.20	GROWTH (10% OF ABOVE)									68		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	748		
TOTALS FROM PAGE ____			
SUBTOTALS			
SYSTEM TOTAL	748		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 14.0

FOR SYSTEM PERSONNEL PROVISIONS

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
14.1	ACCOMMODATIONS FOR PERSONNEL									250		
	SEATS, SUPTS & RESTRAINTS										250	
14.5	EMERGENCY EQUIPMENT									20		
14.20	GROWTH (10% OF ABOVE)									27		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	297		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	297		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 15.0

FOR SYSTEM CREW STATION CONTROLS & PANELS

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
15.4	CREW STATION CONTROLS									280		
	CONTROLS, PANELS & SUPTS										280	
15.20	GROWTH (10% OF ABOVE)									28		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	308		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	308		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 16.0

FOR SYSTEM RANGE SAFETY AND ABORT

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
16.1	RANGE SAFETY & ABORT									250		
16.20	GROWTH (10% OF ABOVE)									25		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	275		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	275		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 17.0

FOR SYSTEM PERSONNEL

FOR ITEM OR MODULE BOOSTER

		UNITS							TOTALS			
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
17.1	CREW									450		
	CREW										400	
	PRESSURE SUITS										50	
17.3	LIFE SUPPORT									16		
17.4	CREW ACCESSORIES									10		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	476		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	476		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 21.0

FOR SYSTEM RESIDUAL PROPELLANT & SERVICE ITEMS

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	**DESCRIPTION**	CLASS (E, C, A)	VOLUME FT³ EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
21.3	FUEL TRAPPED - MAIN ENGINE									3237		
	TANK										2284	
	ENGINE & LINES										953	
21.4	OXIDIZER TRAPPED - MAIN ENGINE									10742		
	TANK										4548	
	ENGINE & LINES										6194	
21.13	SERVICE ITEMS TRAPPED									130		

NOTES:

PAGE TOTALS	14109		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	14109		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 23.0

FOR SYSTEM INFLIGHT LOSSES

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT^3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
23.5	FUEL - ELECTRICAL SYSTEM									16		
23.6	OXIDIZER - ELECTRICAL SYSTEM									143		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	159		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	159		

MSC Form 1522 (Jul 69)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 24.0

FOR SYSTEM THRUST DECAY PROPELLANT

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
24.1	FUEL									31		
24.2	OXIDIZER									198		

NOTES:

	FIRST	SECOND	THIRD
PAGE TOTALS	229		
TOTALS FROM PAGE			
SUBTOTALS			
SYSTEM TOTAL	229		

MSC Form 1522 (Jul 68)

SPACECRAFT DETAIL WEIGHT STATEMENT

PAGE 1 OF 1

CONFIGURATION FR-4 POINT DESIGN (15' x 60' P/L BAY)

BY

DATE

Reference MIL-M-38310A or SP-6004

FOR CODE 25.0

FOR SYSTEM FULL THRUST PROPELLANT

FOR ITEM OR MODULE BOOSTER

		UNITS								TOTALS		
						LOCATION EACH ORBIT REF. SYSTEM				WEIGHT CODE GENERATION		
CODE	DESCRIPTION	CLASS (E, C, A)	VOLUME FT3 EACH	POWER WATTS EACH	WEIGHT LB EACH	X	Y	Z	QUANTITY	FIRST	SECOND	THIRD
25.1	FUEL									201189		
25.2	OXIDIZER									1307726		
25.4	JET FUEL - AIR BREATHING ENGINES									30711		
						PAGE TOTALS				1539626		
						TOTALS FROM PAGE						
						SUBTOTALS						
						SYSTEM TOTAL				1539626		

NOTES:

MSC Form 1522 (Jul 69)

GENERAL DYNAMICS

Convair Division